[redacted] SECURITY INFORMATION

[redacted]

Copy 204
RM L53E07a

[redacted]

NACA

RESEARCH MEMORANDUM

EFFECTS OF A SERIES OF INBOARD PLAN-FORM MODIFICATIONS ON THE LONGITUDINAL CHARACTERISTICS OF TWO 47° SWEPTBACK WINGS OF ASPECT RATIO 3.5, TAPER RATIO 0.2, AND DIFFERENT THICKNESS DISTRIBUTIONS AT MACH NUMBERS OF 1.61 AND 2.01

By Morton Cooper and John R. Sevier, Jr.

Langley Aeronautical Laboratory
Langley Field, Va.

[redacted]

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS
WASHINGTON
July 17, 1953

[redacted]

[redacted]

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUM

EFFECTS OF A SERIES OF INBOARD PLAN-FORM MODIFICATIONS ON THE LONGITUDINAL CHARACTERISTICS OF TWO 47° SWEPTBACK WINGS OF ASPECT RATIO 3.5, TAPER RATIO 0.2, AND DIFFERENT THICKNESS DISTRIBUTIONS AT MACH NUMBERS OF 1.61 AND 2.01

By Morton Cooper and John R. Sevier, Jr.

SUMMARY

Tests of a series of inboard plan-form modifications to two 47° sweptback wings of aspect ratio 3.5 and taper ratio 0.2 were conducted in the Langley 4- by 4-foot supersonic pressure tunnel at Mach numbers of 1.61 and 2.01. One wing had 6-percent-thick hexagonal airfoil sections of constant thickness ratio along the span; the other wing had the same 6-percent-thick sections outboard of the 40-percent-semispan station, but the section thickness increased linearly to 12 percent at the model center line. Inboard plan-form modifications were made by linearly extending the local chord, forward or rearward, from the 40-percent-semispan station to the model center line. Forward or rearward extensions of one-third or two-thirds of the basic center-line chord were tested in various combinations on each wing.

The results indicated that in all cases the addition of the extensions reduced the actual minimum drag (for a given absolute thickness) by an amount which was estimated reasonably well theoretically. Although the lift-curve slopes of the modified wings (when based on wing areas, including extensions) were reduced as anticipated, in all cases, a net increase was realized in maximum lift-drag ratio for the extended-chord configurations.

A specific comparison of two wings of 6-percent-chord thickness, that is, the basic 6-percent-thick wing and the 12-percent-thick wing with 1/3 forward and 2/3 rearward extensions, indicated that the extended 12-percent-thick wing had, at a Mach number of 1.61, about 6 percent higher lift-drag ratio and only 4 percent more minimum drag. Similar gains were present at a Mach number of 2.01. These gains are further enhanced by a volume increase of 67 percent for the extended-chord model.

HADC 1901

INTRODUCTION

In the design of aircraft and their components, aerodynamic considerations tempered with practical requirements combine to dictate the final configurations. For example, in a recent design study of a transonic bomber (some contemplated wing configurations are presented in references 1 and 2), it was found experimentally that increasing the wing volume by increasing the inboard section thickness ratios could be accomplished without subsonic penalties in minimum drag or maximum lift-drag ratio. To be specific, a comparison was made of the aerodynamic characteristics of two wings of identical plan form (sweepback of quarter-chord line 47°, taper ratio 0.2, and aspect ratio 3.5), one wing having 6-percent-thick airfoil sections and the other wing having the 6-percent-thick sections outboard of the 40-percent-semispan station but with section thickness linearly increasing to 12 percent at the model center line. The results indicated that no penalty was incurred in maximum lift-drag ratio (reference 2) up to a Mach number of 0.88 for the thicker wing in spite of its 25 percent greater volume. At the supersonic speeds (refs. 2, 3, and 4), however, the effect of the larger wave drag of the thicker wing (a quantity which was accurately estimated by a strip theory) was clearly evident in reduced lift-drag ratios. Because of the practical advantages of the thicker inboard sections, a further investigation of this type of wing was considered warranted at supersonic speeds in an attempt to maintain its advantages at these speeds.

Since the primary difficulty of the thicker wing was associated with its increased inboard thickness ratio and the consequent greater wave drag at supersonic speeds, two wing models were constructed whereby it was possible to increase the inboard chords and thereby to decrease the local thickness ratios. The two basic wings were identical to those previously tested in references 2 to 4 except that, for construction simplicity, symmetrical hexagonal sections were used. For each wing, one-third of the local chord was removable, forward or rearward, in the inboard 40 percent of the wing semispan. Insert extensions of one-third or two-thirds of the basic center-line root chord were provided and each wing was tested with various combinations of forward and rearward extensions. For all configurations, the extensions increased the inboard chords without changing the wing thickness; the extensions therefore reduced the local thickness ratios from that of the basic wing in all cases.

The purpose of the present paper is to present the aerodynamic characteristics in pitch of these wing configurations for angles of attack up to approximately 8°. The tests were conducted principally at Reynolds numbers of 2.68×10^6 and 2.20×10^6 (based on the mean aerodynamic chord of the basic wing), and for Mach numbers of 1.61 and 2.01, respectively.

SYMBOLS

Free-stream conditions:

M	Mach number
q	dynamic pressure

Wing geometry:

S	area extended through the fuselage
b	span
A	aspect ratio, b^2/S
c	airfoil chord at any spanwise station
y	spanwise distance measured from the plane of symmetry of the wing
$\bar{c}$	mean aerodynamic chord, $\frac{2}{S}\int_0^{b/2} c^2 dy$
α	angle of attack

Force data:

L	lift
D	drag
C_L	lift coefficient, L/qS
$C_{L_{opt}}$	lift coefficient at maximum lift-drag ratio
C_D	drag coefficient, D/qS
$C_{D_{min}}$	minimum drag coefficient
C_m	pitching-moment coefficient about a line perpendicular to plane of symmetry and passing through 25-percent position of mean aerodynamic chord
c.p.	center of pressure

C_{L_α} lift-curve slope, per deg or per radian

C_{m_α} pitching-moment-curve slope

WING DESIGNATION

In order to identify the wing configurations tested, a three-unit numbering system has been adopted, each unit being separated from the others by a dash. The first number (6 or 12) designates the center-line thickness in percent chord of the basic swept wing; the second number (0, 33, or 67) designates the percentage by which the basic center-line chord is extended by the forward insert; and the third number (0, 33, or 67) designates the percentage by which the basic center-line chord is extended by the rearward insert. Thus, the designation 6-0-0 refers to the basic 6-percent-thick wing; whereas the designation 12-33-67 refers to the 12-percent-thick basic wing having a 33 percent forward and a 67 percent rearward extension at the root. In cases where a given number is variable, the number will be replaced by an X. Thus, when curves are plotted as a function of leading-edge extension, the designation will be 6-X-67.

APPARATUS

Tunnel

The tests were conducted in the Langley 4- by 4-foot supersonic pressure tunnel which is a rectangular, closed-throat, single-return wind tunnel designed for a nominal Mach number range from 1.2 to 2.2. The test-section Mach number is varied by deflecting horizontal flexible walls against a series of fixed interchangeable templates which have been designed to produce uniform flow in the test section. For the present investigation the test section Mach numbers were 1.61 and 2.01; the test section heights were 4.4 feet and 5.1 feet, respectively; and the tunnel width was 4.5 feet.

Model

The test model consisted of either of two swept wings (6-X-X or 12-X-X) mounted on an ogive cylinder fuselage (fig. 1) which housed an internal strain-gage balance. The model was sting supported as indicated in figure 1. The angle of attack was measured optically during each test and was varied by rotating the model about the balance moment center.

Wings.- The wings were constructed as indicated in figure 2. Outboard of the 40-percent-semispan station, both wings were constructed of steel and had 6-percent-thick 1/3-1/3-1/3 symmetrical hexagonal airfoil sections (fig. 1). Inboard of the 40-percent-semispan station, the two parallel sides of the hexagonal section (fig. 2) were extended to the side of the fuselage. The airfoil sections in this inboard 40 percent of the semispan were completed by the addition of any combination of the forward and rearward inserts shown in figure 2. Each of the inserts increased the basic center-line chord of the X-0-0 wing by a percentage specified by its designation. Thus, an extension designated 33 (forward or rearward) increased the center-line chord of the basic wing by 33 percent. When the same extensions of the same designation were added forward and rearward, the airfoil section remained at 1/3-1/3-1/3 hexagon (fig. 1).

Two sets of wings and insert sections were tested. One wing with basic inserts (X-0-0) had the 6-percent-thick hexagonal sections extended to the fuselage and thus is designated the 6-0-0 wing. The second wing with basic inserts was identical in plan form, but had linearly increasing airfoil thicknesses from 6 percent at the 40-percent-semispan station to 12 percent at the fuselage center line thereby forming the 12-0-0 wing. Since each of the 6-X-X and 12-X-X wings could be tested with 9 combinations of inserts, there were a total of 18 wing configurations.

Figure 3(a) shows the basic 6-percent-thick wing (6-0-0) and figure 3(b) shows the 6-percent-thick wing with the 33 percent forward and 67 percent rearward extensions (6-33-67).

Fuselage.- The fuselage was an ogive cylinder combination (fig. 1), the ogive having a fineness ratio of 3.5. A six-component strain-gage balance (ref. 5) was housed within the fuselage. For this investigation, only normal force, chord force, and pitching moment were analyzed.

TESTS

Basic data.- All wing configurations shown in figure 4 were tested at a Mach number of 1.61 through an angle-of-attack range from about -2° to 8°. The Reynolds number (based on the mean aerodynamic chord of the X-0-0 wing) was, with the exception of several isolated test conditions, 2.68×10^{6}. For several of the configurations with the larger extensions it was necessary to reduce the stagnation pressure to prevent overloading the balance; the Reynolds number for these test conditions was 2.20×10^{6}. A check test of the 6-0-0 wing at both Reynolds numbers, however, indicated no measurable effect of this slight Reynolds number change.

In order to establish Mach number variations, the extreme configurations shown in the corner sketches of figure 4 (those configurations having basic inserts, 67-percent extensions, or a combination of both) were tested at a Mach number of 2.01 and a Reynolds number of 2.20×10^6.

Preliminary tests.- Prior to the start of the main program, several preliminary tests were made on the 12-0-0 and 12-67-33 wings without the sting block, with a $2\frac{1}{2}$- inch-diameter sting block, and with the 3-inch-diameter sting-block shown in figure 1. In all cases the data, when corrected to free-stream static pressure, agreed within the limits of reproducibility. For all data presented, the 3-inch-diameter sting-block was installed since for this condition the correction of the base pressure to free-stream static pressure (which was applied to all data) was a minimum.

THEORETICAL CALCULATIONS

Drag at Zero Lift

The drag at zero lift of each wing-body configuration was calculated as the sum of the individual drags of the body and the exposed wing. Interference between the wing and body was neglected inasmuch as the body is cylindrical in the zone of influence of the wing and hence can experience no pressure drag in this region. Furthermore, the wing operates in a flow field which is essentially uniform. The pressure drag of the body was calculated by means of the linear theory as presented in reference 6 and the skin friction was estimated by the extended Frankl-Voishel method discussed in reference 7. Turbulent skin friction was assumed for the body on the basis of drag measurements made for several Reynolds numbers on a similar body (ref. 3).

The wave drags of the basic wings (X-0-0) were calculated by linear theory by a procedure similar to that outlined in reference 8. The wave drags of the wings having inboard extensions were estimated by a strip-theory calculation - two-dimensional thickness corrections were applied to the basic wing to allow for the thickness changes as the inserts were added. No correction was made for the plan-form change. The skin friction of the wing was assumed, somewhat arbitrarily, to be laminar and was calculated by the method of reference 9.

Lifting Characteristics

The lift-curve slopes of the wing-body combinations were estimated by the method of reference 10. In the application of this method several

simplifying assumptions (as will be discussed) were made to avoid prohibitively lengthy calculations.

Wing lift-curve slope.- The lift-curve slopes of the basic wing (X-0-0) alone were determined from references 11 and 12; it is to be noted that for the Mach number 1.61 case the leading edge of the wing was essentially sonic. Since the basic trailing edge is supersonic at both Mach numbers, the additional lift on the rear inserts (when no forward extensions are present), and hence on the wing, is given by the integration of the theoretical linear pressures over the inserts. In order to estimate the effect of the forward extension (no rearward extensions present), the reverse flow problem (ref. 13) was considered. In this reverse flow, the insert (fig. 4) lies behind a sonic edge for $M = 1.61$; and, in addition, its own trailing edge is subsonic so that it was assumed (for this Mach number) that the loading on the insert was small and could be neglected. Hence, the total lift on the X-X-0 wings is assumed to be the same as that of the X-0-0 wings for a Mach number of 1.61 (that is, the forward insert is ineffective for producing lift). No calculations are presented for $M = 2.01$ for the forward extension condition because this reasoning does not apply. By means of similar reasoning, the effects of combinations of forward and rearward extensions were obtained.

Wing-body lift-curve slope.- In computing the lift-curve slopes of the wing-body combinations, it was assumed that the inboard section of the wing plan form was of primary importance in determining the effective lift carry-over. Hence, the lift carry-over was computed for a wing of zero taper ratio having the same sweep of the leading and trailing edges as given by the insert sections. It is to be noted that for configurations having the basic forward insert, this assumption entails no further approximations than those inherent in reference 10.

Drag due to lift.- Because of the relative sharpness of the wing leading edge and minor role of subsonic leading edges in the present configurations, the drag due to lift was assumed to be given by the component of the normal force in the drag direction.

RESULTS AND DISCUSSION

Basic data (figs. 5 to 10).- The basic lift, drag, and lift-drag ratio data for the 6-X-X wing and the 12-X-X wing are presented as a function of angle of attack in figures 5 and 6, respectively, for Mach numbers of 1.61 and 2.01. In addition, the lift-drag ratios have been plotted as a function of lift coefficient in figures 7 and 8. All the data presented in these figures as well as in succeeding figures are

tabulated in table I. A summary of the individual wing characteristics (such as minimum drag and lift-curve slope) is presented in tables II and III. The predicted reduction in minimum drag coefficient and the increase in maximum lift-drag ratio with the addition of extensions is clearly evident from figures 5 to 8 and table II. These points will become more evident in subsequent summary plots.

The pitching-moment characteristics of the 6-X-X wing and the 12-X-X wing are presented in figures 9 and 10, respectively. Since each insert section was assumed to form a new wing, a new moment center referenced to the quarter-chord point of the mean aerodynamic chord of each wing was used to reduce the data. This referencing leads, in many cases, to the anomalous result (a fact which is most evident for the X-X-0 wings) that the forward extensions increase stability and rearward extensions decrease stability. This is, in reality, an effect resulting from the fact that the moment axis changes more rapidly than the physical center of pressure, as is evident from the center-of-pressure data also presented in figures 9 and 10.

Minimum drag coefficients (fig. 11).- The minimum drag coefficients of all the wing configurations have been correlated as a function of the sum of the forward and rearward extensions in figure 11. This procedure is consistent with the initial theoretical assumption that the extensions would introduce primarily a thickness effect. In figure 11(a) the data have been nondimensionalized in terms of the individual wing areas; whereas in figure 11(b), the area of the X-0-0 wing has been used throughout. Hence, these latter coefficients (fig. 11(b)) are equivalent to direct forces.

Figure 11(a) indicates that the drag results correlate quite well with the thickness-correction concept, deviating primarily for the larger insert combinations as might be anticipated. The experimental data are considerably below the theoretical curves but this is a deficiency of the theory in predicting the basic wing (X-0-0) characteristics rather than in predicting the effects of the extensions on the basic wing characteristics. This is apparent since, when the theoretical curve is adjusted arbitrarily by so shifting the curve that theory and experiment agree for the basic wing, the estimated correlation curve is quite good. Hence, it can be concluded that, for a given basic wing of known characteristics, the effects of inboard plan-form extensions on the drag can be estimated reasonably well.

It is of practical interest to note from figure 11(b) or tables II and III the results for a specific illustrative comparison at a Mach number of 1.61. For example, the 12-33-67 wing can be compared with the 6-0-0 conventional wing, observing, of course, that both wings have 6-percent-thick sections throughout. It is to be noted that the slight difference in airfoil section of the two wings in the inboard region introduces (based on two-dimensional linear theory calculations)

a negligible effect on this minimum drag comparison.

Wing designation	$C_{D_{min}}$	$C_{D_{min_{X-0-0}}}$
6-0-0	0.0238	0.0238
12-33-67	.0186	.0248

Hence, the 12-33-67 wing, while having only 4 percent more minimum drag (a 21-percent lower drag coefficient) than the more conventional 6-0-0 wing, has 67 percent more wing volume, a parameter of the utmost practical importance.

Drag due to lift (figs. 12 and 13).- The drag-due-to-lift parameter is presented in figures 12 and 13 for the 6-X-X wings and 12-X-X wings, respectively, as a function of forward extension for constant values of trailing-edge extension. At a Mach number of 1.61, the results for both wing families are qualitatively the same. In all cases (figs. 12(b) and 13(b)) the drag-due-to-lift parameter is less than the reciprocal of the experimental lift-curve slope (in radians) indicating that the resultant force on the airfoil due to incidence is inclined forward of the normal to the chord. The comparison of the experimental drag-due-to-lift parameter with the reciprocal of the theoretical lift-curve slope (figs. 12(a) and 13(a)) for a Mach number of 1.61 is misleading in the exceptional agreement indicated in view of the results of figures 12(b) and 13(b). This coincidental agreement arises (as will be established) because the theoretical lift-curve slopes are too great and thereby compensate for the forward inclination of the resultant force vector previously mentioned. The data at a Mach number of 2.01 indicate, perhaps, a less forward inclination of the resultant force (possible exception being 12-0-0 and 12-0-67) than at a Mach number of 1.61 but in general are too incomplete to warrant a more positive observation.

Lift-curve slope (figs. 14 and 15).- The lift-curve-slope data (figs. 14(a) and 14(b)) for both the 6-X-X and 12-X-X wing series show considerable overestimation of the experimental results by the theory at a Mach number of 1.61 with a considerably better estimate, at least for the basic wings (X-0-0) at a Mach number of 2.01. The improved agreement at $M = 2.01$ coupled with the fact that the overestimation at $M = 1.61$ is a maximum for the basic leading edge (and all trailing edges) indicates that the main difficulties are, perhaps, associated with the sonic leading edge at $M = 1.61$. The theory ($M = 1.61$), when adjusted to correspond to the experimental data of the basic wing, reasonably estimates the effects of the extensions; however, the discrepancies still remain significant because only small differences are sought in the first place. Regarding the theoretical assumption ($M = 1.61$) that the rearward extension is more effective in producing lift than the forward extensions,

the data of figures 14(b) and 15(b) (or table III) appear to substantiate this contention. However, it must be noted that, theoretically, the wing-body carry-over effect of the rear extension is larger and hence may account for a significant part of the added effectiveness. In any case, although the experimental data are not conclusive as to the validity of the detailed assumptions, the result that the rearward extension is more effective is substantiated even for a Mach number of 2.01.

Maximum lift-drag ratio (figs. 16 and 17).- The data for the maximum lift-drag ratio presented for both wing families in figures 16 and 17 indicate that the adjusted theory quite reasonably predicts the effects of the chord extensions except perhaps for the combinations of large extensions. In all cases, the addition of the extensions improved the maximum lift-drag ratio and reduced the lift coefficient for maximum lift-drag ratio.

To be specific, again compare at a Mach number of 1.61 the same two 6-percent-thick wings discussed previously:

Wing designation	$(L/D)_{max}$	$C_{L_{opt}}$
6-0-0	5.88	0.260
12-33-67	6.29	.213

Here again the advantages of the extended-root-chord wing are evident. The 12-33-67 wing (having 67 percent more volume) has about a 6 percent higher maximum lift-drag ratio occurring at a lower lift coefficient and with only 4 percent more minimum drag. Similar gains would be anticipated at a Mach number of 2.01. This increase in maximum lift-drag ratio appears to be a plan-form effect rather than a Reynolds number effect on skin friction (associated with the extended chord of the 12-33-67 wing) since calculations made on the assumption of turbulent flow on the wings show no material effect on the comparison.

Two additional points of general interest remain to be noted. The first is that these data were obtained from relatively crude models designed to facilitate the testing of various arrangements. The results, therefore, are to be applied more for indicating trends than for the specific numbers presented since, with the use of better airfoil sections, improvements in maximum lift-drag ratio could be realized. Secondly, these data were obtained solely for supersonic speeds, and hence, in the absence of transonic data, no definite conclusions can be drawn concerning the possible application of these ideas to configurations which may be designed primarily for transonic use with short periods of supersonic flight.

Pitching-moment-curve slopes (figs. 18 and 19).- The pitching-moment-curve slopes (figs. 18 and 19) reflect the difficulty (mentioned previously) in treating each configuration as a separate wing and in relocating the moment axis for each wing. The centers of pressure for a representative angle of attack, however, show the anticipated rearward shift with the addition of rearward extensions and the forward shift with the addition of forward extensions.

CONCLUDING REMARKS

Tests of a series of inboard plan-form modifications to two 47° sweptback wings of aspect ratio 3.5 and taper ratio 0.2 were conducted in the Langley 4- by 4-foot supersonic pressure tunnel at Mach numbers of 1.61 and 2.01. One wing had 6-percent-thick hexagonal airfoil sections of constant thickness ratio along the span; the other wing had the same sections outboard of the 40-percent-semispan station but with thickness linearly increasing to 12 percent at the model center line. Inboard plan-form modifications were made by linearly extending the local chord, forward or rearward, from the 40-percent-semispan station to the model center line. Forward or rearward extensions of one-third or two-thirds of the basic center-line chord were tested in various combinations on each wing.

The results indicated that, in all cases, the addition of the extensions reduced the actual minimum drag (for a given absolute thickness) by an amount which was estimated by theory reasonably well. Although the lift-curve slopes of the modified wings (when based on wing areas, including extensions) were reduced as anticipated, there was, in all cases, a net increase in maximum lift-drag ratio for the extended-chord configurations.

A specific comparison of two wings of 6-percent thickness, that is, the basic 6-percent-thick wing and the 12-percent-thick wing with 1/3 forward and 2/3 rearward extensions, indicated that the extended 12-percent-thick wing had, at a Mach number of 1.61, about 6 percent higher lift-drag ratio and only 4 percent more minimum drag. Similar gains were present at a Mach number of 2.01. These gains are further enhanced by a volume increase of 67 percent for the extended-chord model.

Langley Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., May 14, 1953.

REFERENCES

1. Harris, William G.: A Wind-Tunnel Investigation at High-Subsonic and Low-Supersonic Mach Numbers on a Series of Wings with Various Sweepback, Taper, Aspect Ratio, and Thickness. AF Tech. Rep. No. 6669, pt. I, Wright Air Dev. Center, U. S. Air Force, Oct. 1951.

2. Bielat, Ralph P., Harrison, Daniel E., and Coppolino, Domenic A.: An Investigation at Transonic Speeds of the Effects of Thickness Ratio and of Thickened Root Stations on the Aerodynamic Characteristics of Wings with 47° Sweepback, Aspect Ratio 3.5, and Taper Ratio 0.2 in the Slotted Test Section of the Langley 8-Foot High-Speed Tunnel. NACA RM L51I04a, 1951.

3. Robinson, Ross B., and Driver, Cornelius: Aerodynamic Characteristics at Supersonic Speeds of a Series of Wing-Body Combinations Having Cambered Wings with an Aspect Ratio of 3.5 and Taper Ratio of 0.2. Effects of Sweep Angle and Thickness Ratio on the Aerodynamic Characteristics in Pitch at $M = 1.60$. NACA RM L51K16a, 1952.

4. Robinson, Ross B.: Aerodynamic Characteristics at Supersonic Speeds of a Series of Wing-Body Combinations Having Cambered Wings With an Aspect Ratio of 3.5 and a Taper Ratio of 0.2. Effects of Sweep Angle and Thickness Ratio on the Aerodynamic Characteristics in Pitch at $M = 2.01$. NACA RM L52E09, 1952.

5. Robinson, Ross B.: An Investigation of a Supersonic Aircraft Configuration Having a Tapered Wing With Circular-Arc Sections and 40° Sweepback. Static Lateral Control Characteristics at Mach Numbers of 1.40 and 1.59. NACA RM L50I11, 1950.

6. Lighthill, M. J.: Supersonic Flow Past Bodies of Revolution. R. & M. No. 2003, British A.R.C., 1945.

7. Rubesin, Morris W., Maydew, Randall C., and Varga, Steven A.: An Analytical and Experimental Investigation of the Skin Friction of the Turbulent Boundary Layer on a Flat Plate at Supersonic Speeds. NACA TN 2305, 1951.

8. Beane, Beverly: The Characteristics of Supersonic Wings Having Biconvex Sections. Jour. Aero. Sci., vol. 18, no. 1, Jan. 1951, pp. 7-20.

9. Chapman, Dean R., and Rubesin, Morris W.: Temperature and Velocity Profiles in the Compressible Laminar Boundary Layer With Arbitrary Distribution of Surface Temperature. Jour. Aero. Sci., vol. 16, no. 9, Sept. 1949, pp. 547-565.

10. Nielsen, Jack N., and Kaattari, George E.: Method for Estimating Lift Interference of Wing-Body Combinations at Supersonic Speeds. NACA RM A51J04, 1951.

11. Malvestuto, Frank S., Margolis, Kenneth, and Ribner, Herbert S.: Theoretical Lift and Damping in Roll at Supersonic Speeds of Thin Sweptback Wings With Streamwise Tips, Subsonic Leading Edges, and Supersonic Trailing Edges. NACA Rep. 970, 1950. (Supersedes NACA TN 1860.)

12. Harmon, Sidney M., and Jeffreys, Isabella: Theoretical Lift and Damping in Roll of Thin Wings with Arbitrary Sweep and Taper at Supersonic Speeds. Supersonic Leading and Trailing Edges. NACA TN 2114, 1950.

13. Brown, Clinton E.: The Reversibility Theorem for Thin Airfoils in Subsonic and Supersonic Flow. NACA Rep. 986, 1950. (Supersedes NACA TN 1944.)

TABLE I.- BASIC DATA

M = 1.61					M = 2.01				
α, deg	C_L	C_D	L/D	C_m	α, deg	C_L	C_D	L/D	C_m
Wing 6-0-0									
0.13	0.008	0.0238	0.34	-0.0021	-0.07	0.002	0.0213	0.07	0.0000
-1.85	-.088	.0262	-3.36	.0147	1.88	.083	.0244	3.40	-.0137
4.12	.210	.0369	5.68	-.0415	3.08	.131	.0284	4.60	-.0215
6.07	.306	.0527	5.81	-.0615	4.18	.173	.0340	5.08	-.0279
8.02	.400	.0753	5.32	-.0789	5.08	.207	.0397	5.21	-.0331
10.02	.488	.1047	4.66	-.0909	5.97	.240	.0465	5.17	-.0378
7.08	.354	.0635	5.57	-.0707	6.88	.278	.0549	5.06	-.0426
5.13	.260	.0442	5.87	-.0519	8.25	.329	.0695	4.73	-.0483
2.15	.107	.0268	3.99	-.0194	9.22	.361	.0805	4.49	-.0514
.20	.010	.0238	.43	-.0022	-1.85	-.076	.0240	-3.16	.0125
					.03	.002	.0219	.11	-.0002
Wing 6-33-0									
0.12	0.007	0.0206	0.36	-0.0029					
2.13	.100	.0232	4.32	-.0303					
4.22	.199	.0333	5.98	-.0615					
6.23	.295	.0501	5.89	-.0903					
8.28	.382	.0735	5.20	-.1137					
7.28	.340	.0611	5.56	-.1028					
5.28	.249	.0411	6.06	-.0766					
-1.92	-.087	.0230	-3.76	.0246					
.13	.007	.0205	.33	-.0026					
Wing 6-67-0									
0.08	0.006	0.0186	0.31	-0.0030	0.02	0.002	0.0166	0.11	-0.0010
-2.15	-.087	.0212	-4.11	.0301	2.35	.085	.0202	4.22	-.0288
2.15	.092	.0210	4.37	-.0337	4.18	.151	.0276	5.48	-.0496
4.30	.190	.0314	6.06	-.0700	5.32	.190	.0345	5.50	-.0613
5.30	.237	.0392	6.03	-.0863	3.25	.117	.0232	5.06	-.0391
8.05	.344	.0661	5.20	-.1219	-2.13	-.076	.0192	-3.97	.0249
7.07	.306	.0552	5.54	-.1103	7.57	.260	.0518	5.02	-.0813
6.12	.266	.0455	5.85	-.0968	6.80	.243	.0459	5.29	-.0767
1.17	.049	.0192	2.53	-.0182	6.15	.214	.0395	5.42	-.0682
.15	.008	.0186	.44	-.0038	.02	.002	.0165	.13	-.0012

TABLE I.- BASIC DATA - Continued

M = 1.61					M = 2.01				
α, deg	C_L	C_D	L/D	C_m	α, deg	C_L	C_D	L/D	C_m
Wing 6-0-33									
0.05	0.005	0.0212	0.24	-0.0017					
-2.00	-.090	.0236	-3.83	.0231					
1.97	.095	.0232	4.11	-.0252					
4.00	.195	.0322	6.05	-.0536					
5.93	.288	.0468	6.15	-.0804					
7.87	.377	.0673	5.60	-.1046					
6.88	.333	.0562	5.93	-.0929					
4.98	.242	.0388	6.25	-.0673					
3.08	.150	.0274	5.47	-.0406					
1.02	.051	.0216	2.34	-.0133					
.08	.006	.0211	.27	-.0017					
Wing 6-33-33									
0.10	0.006	0.0185	0.34	-0.0024					
-1.97	-.086	.0215	-4.00	.0257					
2.15	.096	.0212	4.54	-.0298					
4.10	.188	.0299	6.28	-.0592					
6.12	.278	.0449	6.19	-.0872					
8.15	.363	.0664	5.46	-.1117					
7.08	.322	.0547	5.88	-.1001					
5.15	.234	.0368	6.36	-.0736					
2.15	.096	.0211	4.55	-.0274					
3.13	.141	.0247	5.70	-.0440					
.07	.005	.0184	.29	-.0020					
Wing 6-67-33									
0.05	0.004	0.0169	0.26	-0.0021					
-2.00	-.081	.0192	-4.20	.0270					
2.15	.090	.0193	4.66	-.0314					
3.15	.134	.0230	5.81	-.0468					
4.23	.180	.0285	6.33	-.0630					
5.27	.223	.0355	6.28	-.0774					
6.32	.267	.0445	5.99	-.0920					
7.22	.304	.0535	5.68	-.1039					
.13	.007	.0169	.41	-.0029					

NACA

TABLE I.- BASIC DATA - Continued

M = 1.61					M = 2.01				
α, deg	C_L	C_D	L/D	C_m	α, deg	C_L	C_D	L/D	C_m
Wing 6-0-67									
0.10	0.005	0.0192	0.25	-0.0012	0.00	0.001	0.0171	0.07	0.0000
-1.85	-.083	.0210	-3.95	.0206	2.12	.080	.0203	3.96	-.0191
2.05	.093	.0212	4.38	-.0231	3.02	.114	.0229	4.96	-.0271
4.03	.184	.0295	6.23	-.0477	3.95	.148	.0268	5.51	-.0352
4.92	.227	.0351	6.46	-.0593	4.98	.185	.0324	5.71	-.0440
5.88	.270	.0425	6.36	-.0710	5.97	.219	.0389	5.62	-.0518
6.82	.311	.0509	6.11	-.0819	7.00	.256	.0472	5.43	-.0602
7.80	.354	.0610	5.80	-.0929	8.10	.294	.0572	5.14	-.0684
3.08	.141	.0247	5.71	-.0359	-1.98	-.075	.0197	-3.80	.0183
.15	.007	.0191	.38	-.0017	.07	.002	.0170	.14	-.0003
Wing 6-33-67									
0.12	0.005	0.0166	0.30	-0.0016					
-1.90	-.080	.0189	-4.23	.0224					
2.10	.089	.0190	4.67	-.0254					
3.17	.135	.0225	6.01	-.0391					
4.10	.178	.0272	6.55	-.0517					
5.08	.221	.0333	6.63	-.0642					
6.08	.264	.0410	6.43	-.0763					
7.05	.304	.0497	6.12	-.0877					
.15	.007	.0166	.42	-.0021					
Wing 6-67-67									
0.17	0.007	0.0153	0.44	-0.0025	0.00	-0.002	0.0144	-0.14	0.0006
2.20	.088	.0177	4.97	-.0281	2.20	.076	.0169	4.51	-.0228
3.18	.128	.0210	6.11	-.0411	3.98	.137	.0231	5.94	-.0402
4.23	.172	.0261	6.59	-.0550	6.90	.233	.0420	5.55	-.0667
5.25	.213	.0326	6.54	-.0678	8.02	.268	.0519	5.17	-.0762
6.28	.254	.0406	6.25	-.0802	7.40	.251	.0465	5.39	-.0714
7.30	.293	.0500	5.86	-.0920	6.02	.206	.0354	5.81	-.0591
8.33	.333	.0611	5.45	-.1038	5.02	.173	.0288	6.02	-.0503
-1.98	-.078	.0174	-4.47	.0241	3.02	.107	.0194	5.54	-.0317
.12	.005	.0153	.31	-.0019	-1.98	-.072	.0165	-4.35	.0212
					.00	.001	.0145	.05	-.0002

NACA

TABLE I.- BASIC DATA - Continued

M = 1.61					M = 2.01				
α, deg	C_L	C_D	L/D	C_m	α, deg	C_L	C_D	L/D	C_m
Wing 12-0-0									
0.13	0.006	0.0297	0.20	-0.0018	0.03	0.002	0.0266	0.08	-0.0004
2.05	.095	.0323	2.95	-.0170	2.22	.090	.0297	3.04	-.0139
4.13	.197	.0414	4.75	-.0369	3.05	.122	.0323	3.77	-.0186
6.03	.293	.0562	5.21	-.0563	4.05	.162	.0370	4.38	-.0245
8.05	.384	.0782	4.91	-.0734	5.12	.202	.0434	4.65	-.0299
8.98	.430	.0911	4.71	-.0804	6.10	.240	.0511	4.69	-.0349
7.07	.339	.0666	5.08	-.0653	7.18	.281	.0608	4.62	-.0398
5.15	.246	.0483	5.09	-.0469	8.10	.315	.0702	4.48	-.0434
3.22	.151	.0366	4.12	-.0271	-2.08	-.084	.0287	-2.92	.0126
-1.88	-.087	.0320	-2.73	.0139	.03	.001	.0267	.04	-.0003
.15	.005	.0296	.18	-.0017					
Wing 12-33-0									
0.13	0.006	0.0252	0.24	-0.0026					
2.13	.094	.0277	3.38	-.0280					
4.17	.188	.0363	5.18	-.0624					
6.23	.284	.0528	5.37	-.0864					
8.30	.374	.0762	4.91	-.1115					
7.28	.329	.0636	5.18	-.0995					
5.27	.239	.0446	5.37	-.0732					
3.30	.147	.0321	4.57	-.0443					
-1.90	-.084	.0273	-3.08	.0235					
.12	.005	.0251	.22	-.0023					
Wing 12-67-0									
0.12	0.007	0.0216	0.31	-0.0029	0.03	0.001	0.0191	0.06	-0.0004
2.10	.089	.0248	3.57	-.0323	2.37	.083	.0221	3.77	-.0276
4.22	.180	.0341	5.29	-.0662	4.20	.147	.0292	5.02	-.0474
6.25	.267	.0499	5.34	-.0963	5.28	.184	.0359	5.14	-.0590
7.22	.307	.0593	5.17	-.1096	6.35	.219	.0435	5.03	-.0691
5.23	.223	.0410	5.44	-.0813	-2.17	-.077	.0215	-3.56	.0251
3.23	.136	.0286	4.76	-.0498	7.25	.243	.0492	4.95	-.0753
-1.85	-.075	.0239	-3.15	.0263	8.15	.272	.0576	4.72	-.0834
.12	.007	.0222	.30	-.0029					

NACA

TABLE I.- BASIC DATA - Continued

M = 1.61					M = 2.01				
α, deg	C_L	C_D	L/D	C_m	α, deg	C_L	C_D	L/D	C_m
Wing 12-0-33									
0.12	0.007	0.0253	0.27	-0.0021					
-1.92	-.084	.0277	-3.03	.0211					
2.07	.093	.0278	3.35	-.0243					
3.05	.140	.0311	4.49	-.0365					
4.03	.187	.0360	5.20	-.0502					
5.02	.236	.0425	5.55	-.0637					
5.98	.280	.0501	5.59	-.0768					
6.97	.325	.0595	5.47	-.0896					
7.90	.371	.0701	5.29	-.1019					
.17	.007	.0252	.26	-.0019					
Wing 12-33-33									
0.10	0.006	0.0211	0.27	-0.0021					
-1.83	-.078	.0232	-3.34	.0230					
2.12	.090	.0235	3.82	-.0275					
3.23	.140	.0274	5.11	-.0435					
4.20	.184	.0323	5.69	-.0576					
5.18	.229	.0388	5.89	-.0718					
6.17	.274	.0470	5.83	-.0858					
7.17	.315	.0565	5.57	-.0982					
.15	.007	.0211	.35	-.0024					
Wing 12-67-33									
0.08	0.005	0.0187	0.25	-0.0018					
2.10	.086	.0209	4.10	-.0298					
4.27	.178	.0299	5.94	-.0620					
6.37	.264	.0456	5.79	-.0909					
-1.93	-.078	.0209	-3.71	.0264					
.12	.005	.0186	.25	-.0018					
3.12	.128	.0241	5.31	-.0447					
5.13	.214	.0356	6.00	-.0744					
8.10	.336	.0644	5.22	-.1140					

NACA

TABLE I.- BASIC DATA - Concluded

M = 1.61					M = 2.01				
α, deg	C_L	C_D	L/D	C_m	α, deg	C_L	C_D	L/D	C_m
Wing 12-0-67									
0.17	0.006	0.0223	0.25	-0.0013	0.05	0.001	0.0206	0.04	0.0000
2.02	.086	.0245	3.49	-.0209	2.15	.078	.0229	3.41	-.0180
4.00	.177	.0320	5.53	-.0445	3.03	.110	.0253	4.34	-.0255
5.92	.264	.0449	5.88	-.0679	4.00	.145	.0293	4.94	-.0335
7.98	.348	.0639	5.45	-.0901	5.03	.182	.0347	5.23	-.0419
6.85	.305	.0530	5.76	-.0790	6.00	.215	.0409	5.26	-.0496
4.98	.221	.0378	5.83	-.0561	7.08	.253	.0494	5.12	-.0578
-1.85	-.080	.0246	-3.24	.0193	7.92	.282	.0569	4.96	-.0641
					-1.98	-.074	.0223	-3.32	.0175
					.03	.001	.0206	.04	.0000
Wing 12-33-67									
0.08	0.004	0.0187	0.21	-0.0011					
2.03	.083	.0207	4.00	-.0232					
4.10	.172	.0284	6.04	-.0493					
5.03	.213	.0339	6.29	-.0615					
6.05	.256	.0414	6.17	-.0737					
6.97	.295	.0496	5.95	-.0847					
8.03	.334	.0599	5.57	-.0951					
1.13	.045	.0191	2.36	-.0126					
-1.90	-.077	.0207	-3.71	.0214					
.10	.004	.0186	.21	-.0011					
Wing 12-67-67									
0.15	0.005	0.0166	0.29	-0.0015	0.02	0.001	0.0151	0.05	0.0001
-1.97	-.076	.0191	-4.00	.0239	2.30	.074	.0178	4.15	-.0217
2.10	.082	.0188	4.35	-.0257	4.12	.134	.0241	5.57	-.0389
3.17	.125	.0222	5.62	-.0395	5.22	.168	.0298	5.64	-.0483
4.22	.168	.0272	6.17	-.0532	6.27	.201	.0366	5.49	-.0573
5.17	.206	.0330	6.24	-.0651	7.43	.236	.0453	5.21	-.0669
6.27	.250	.0413	6.04	-.0783	-2.10	-.070	.0174	-4.02	.0209
.15	.005	.0165	.29	-.0015					

NACA

TABLE II

SUMMARY OF CHARACTERISTICS BASED ON INDIVIDUAL WING AREAS

M = 1.61							M = 2.01						
Wing	$C_{D_{min}}$	$\frac{C_D - C_{D_{min}}}{C_L^2}$	C_{L_α}	$(L/D)_{max}$	$C_{L_{opt}}$	C_{m_α}	Wing	$C_{D_{min}}$	$\frac{C_D - C_{D_{min}}}{C_L^2}$	C_{L_α}	$(L/D)_{max}$	$C_{L_{opt}}$	C_{m_α}
6-0-0	0.0238	0.308	0.0498	5.88	0.260	-0.0090	6-0-0	0.0215	0.430	0.0423	5.20	0.208	-0.0071
6-0-33	.0211	.302	.0478	6.25	.240	-.0123							
6-0-67	.0191	.316	.0455	6.46	.232	-.0117	6-0-67	.0170	.455	.0374	5.70	.186	-.0090
6-33-0	.0205	.335	.0462	6.06	.249	-.0137							
6-33-33	.0184	.340	.0450	6.36	.212	-.0141							
6-33-67	.0166	.349	.0431	6.63	.218	-.0123							
6-67-0	.0186	.364	.0437	6.07	.200	-.0152	6-67-0	.0165	.480	.0355	5.50	.185	-.0121
6-67-33	.0169	.363	.0416	6.33	.190	-.0144							
6-67-67	.0153	.379	.0402	6.60	.185	-.0129	6-67-67	.0143	.479	.0351	6.03	.169	-.0102
12-0-0	.0296	.316	.0468	5.21	.291	-.0082	12-0-0	.0265	.405	.0404	4.70	.240	-.0063
12-0-33	.0252	.317	.0455	5.60	.280	-.0115							
12-0-67	.0223	.323	.0442	5.88	.262	-.0110	12-0-67	.0206	.427	.0367	5.26	.212	-.0087
12-33-0	.0251	.342	.0450	5.37	.262	-.0130							
12-33-33	.0211	.342	.0436	5.89	.232	-.0134							
12-33-67	.0186	.346	.0420	6.29	.213	-.0130							
12-67-0	.0217	.378	.0420	5.44	.230	-.0150	12-67-0	.0192	.488	.0349	5.12	.184	-.0116
12-67-33	.0187	.374	.0412	6.00	.215	-.0142							
12-67-67	.0165	.389	.0397	6.25	.205	-.0136	12-67-67	.0151	.522	.0324	5.64	.168	-.0098

NACA

CONFIDENTIAL

TABLE III

SUMMARY OF CHARACTERISTICS BASED ON AREA OF X-0-0 WING

M = 1.61							M = 2.01						
Wing	$C_{D_{min}}$	$\frac{C_D - C_{D_{min}}}{C_L^2}$	C_{L_α}	$(L/D)_{max}$	$C_{L_{opt}}$	c.p.* at $\alpha = 2^\circ$	Wing	$C_{D_{min}}$	$\frac{C_D - C_{D_{min}}}{C_L^2}$	C_{L_α}	$(L/D)_{max}$	$C_{L_{opt}}$	c.p.* at $\alpha = 2^\circ$
6-0-0	0.0238	0.308	0.0498	5.88	0.260	0.431	6-0-0	0.0215	0.430	0.0423	5.20	0.208	0.411
6-0-33	.0235	.272	.0531	6.25	.267	.470							
6-0-67	.0233	.259	.0556	6.46	.284	.497	6-0-67	.0208	.372	.0457	5.70	.227	.480
6-33-0	.0228	.302	.0513	6.06	.277	.407							
6-33-33	.0225	.278	.0550	6.36	.259	.440							
6-33-67	.0221	.262	.0575	6.63	.291	.470							
6-67-0	.0227	.298	.0534	6.07	.244	.373	6-67-0	.0202	.393	.0434	5.50	.226	.330
6-67-33	.0225	.272	.0555	6.33	.253	.401							
6-67-67	.0221	.262	.0581	6.60	.267	.440	6-67-67	.0207	.332	.0507	6.03	.244	.400
12-0-0	.0296	.316	.0468	5.21	.291	.425	12-0-0	.0265	.405	.0404	4.70	.240	.402
12-0-33	.0280	.285	.0506	5.60	.311	.462							
12-0-67	.0273	.264	.0540	5.88	.320	.490	12-0-67	.0252	.349	.0449	5.26	.259	.470
12-33-0	.0279	.308	.0500	5.37	.291	.402							
12-33-33	.0258	.280	.0533	5.89	.284	.435							
12-33-67	.0248	.260	.0560	6.29	.284	.465							
12-67-0	.0265	.309	.0513	5.44	.281	.373	12-67-0	.0235	.399	.0427	5.12	.225	.328
12-67-33	.0249	.281	.0549	6.00	.287	.405							
12-67-67	.0238	.269	.0573	6.25	.296	.432	12-67-67	.0218	.361	.0468	5.64	.243	.392

*Fraction of $\bar{c}$ of X-0-0 wing.

NACA

CONFIDENTIAL

CONFIDENTIAL

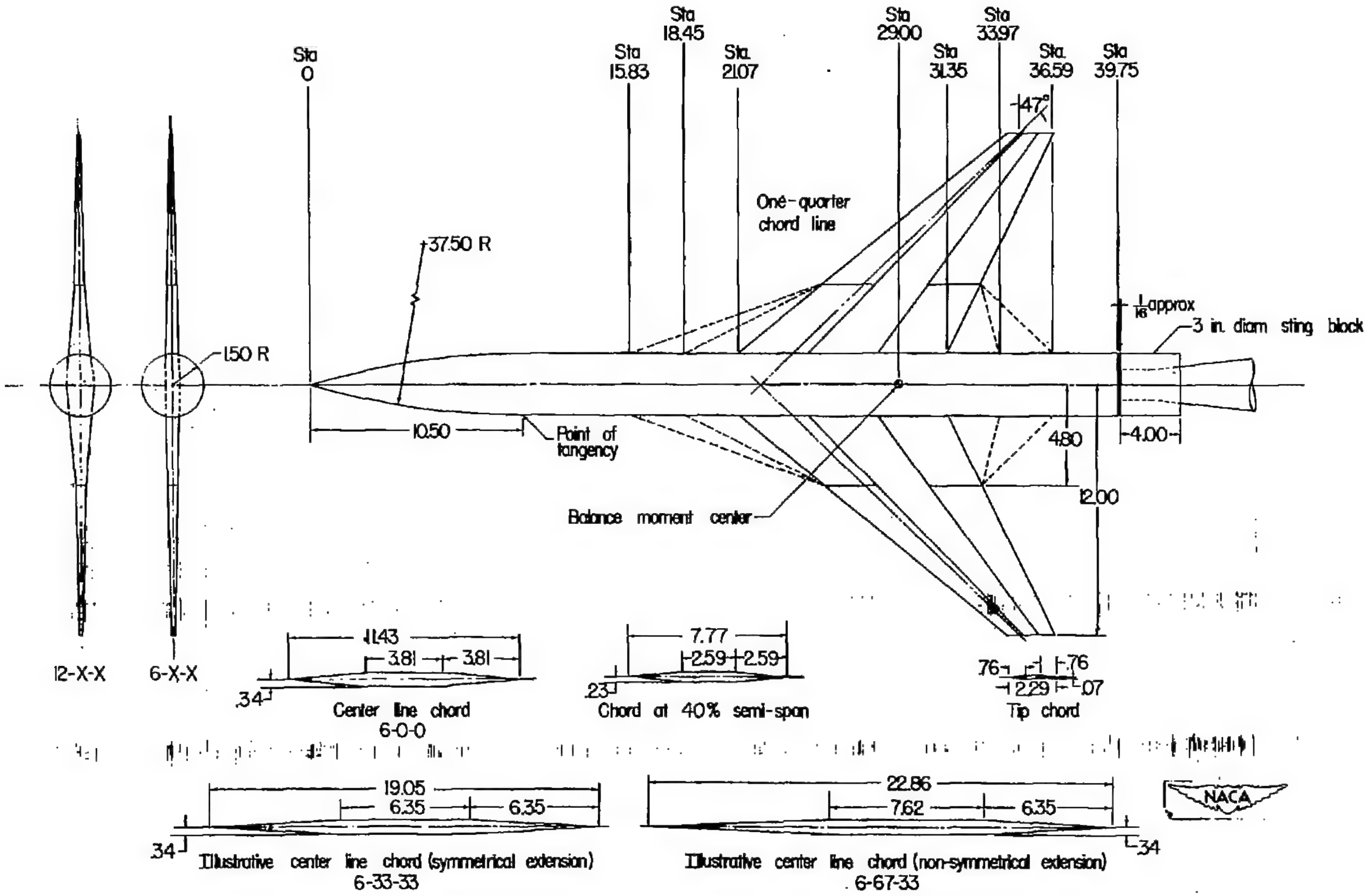


Figure 1.- Schematic layout of model. (All dimensions in inches unless otherwise specified.)

CONFIDENTIAL

CONFIDENTIAL

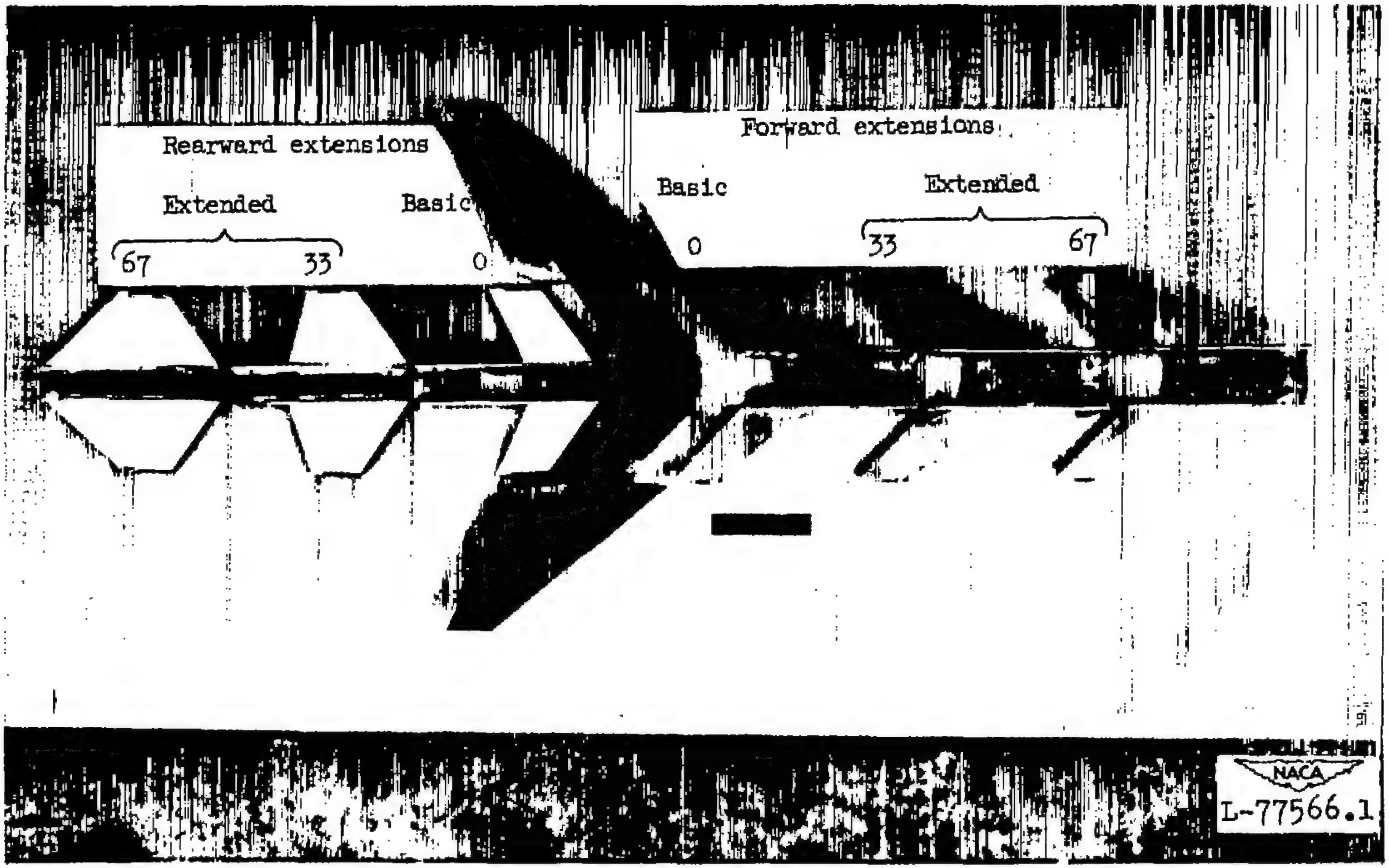
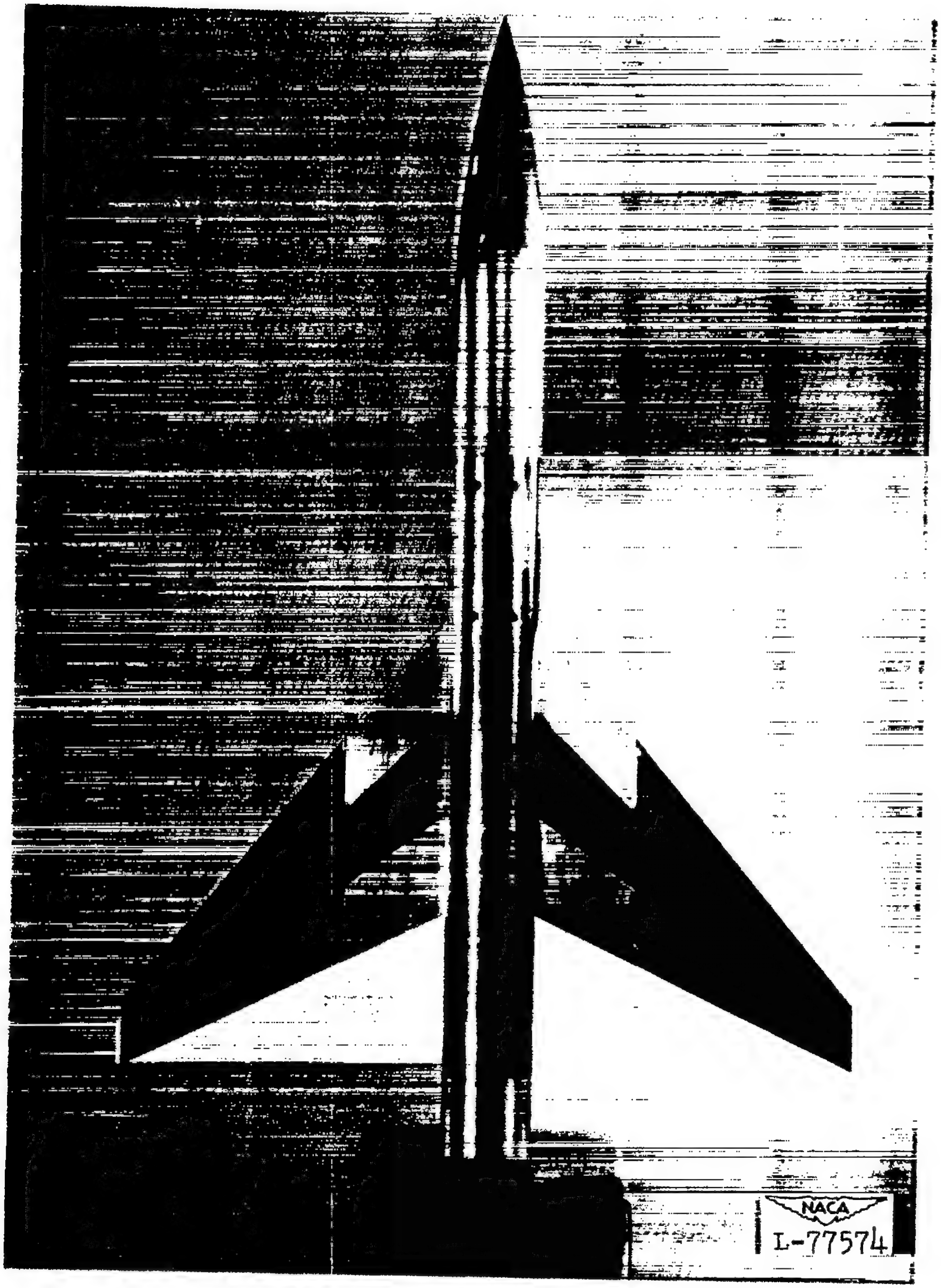


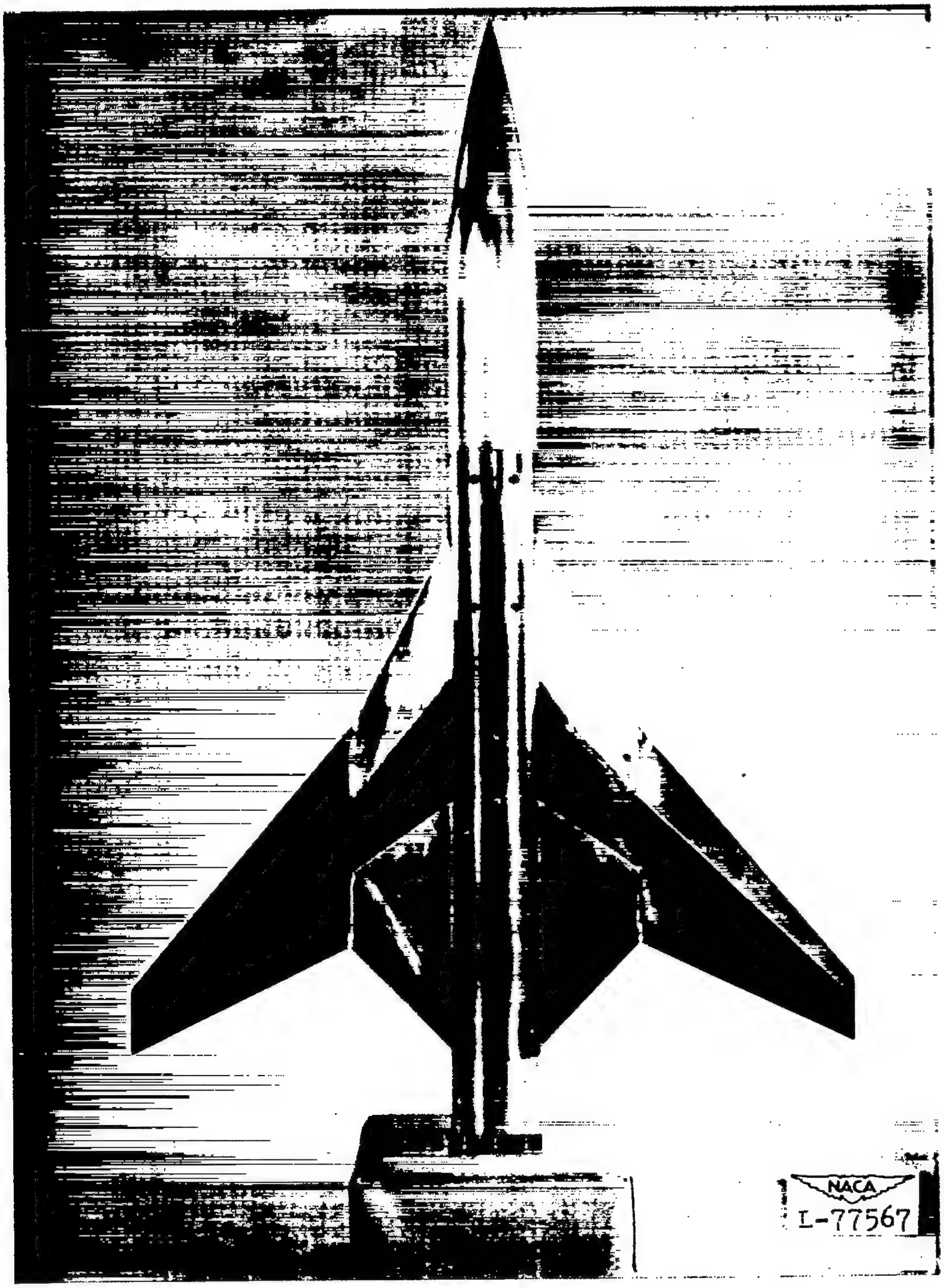
Figure 2.- Wing assembly details.

CONFIDENTIAL



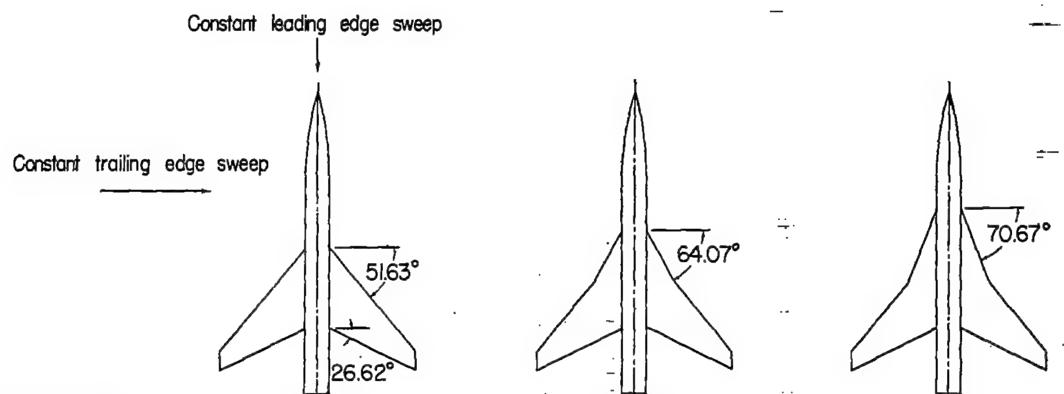
(a) The 6-0-0 wing in combination with body.

Figure 3.- Wing-body configurations.

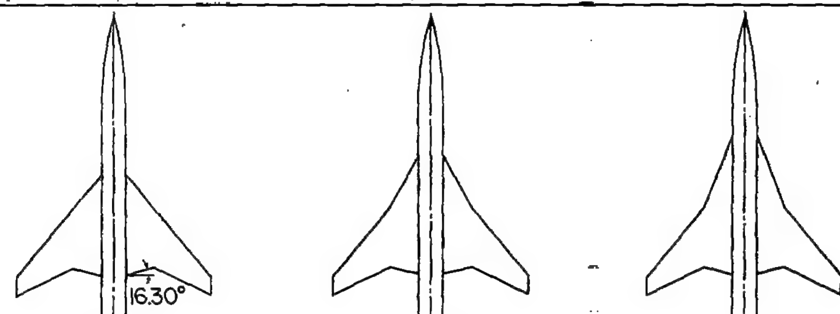


(b) The 6-33-67 wing in combination with body.

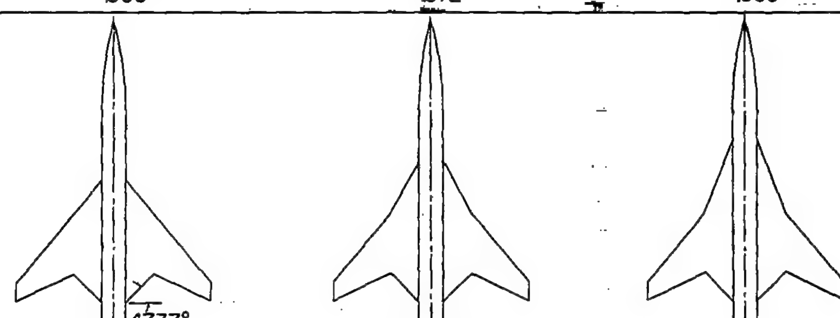
Figure 3.- Concluded.



Wing designation	X-0-0	X-33-0	X-67-0
Area, sq ft	1.143	1.270	1.397
Taper ratio	.200	.150	.120
Mean aerodynamic chord, ft	.656	.782	.923
Aspect ratio	3.500	3.150	2.860
Center line thickness ratio:			
6-X-0	.060	.045	.036
12-X-0	.120	.090	.072



Wing designation	X-0-33	X-33-33	X-67-33
Area, sq ft	1.270	1.397	1.524
Taper ratio	.150	.120	.100
Mean aerodynamic chord, ft	.782	.923	1.077
Aspect ratio	3.150	2.860	2.620
Center line thickness ratio:			
6-X-33	.045	.036	.030
12-X-33	.090	.072	.060



Wing designation	X-0-67	X-33-67	X-67-67
Area, sq ft	1.397	1.524	1.651
Taper ratio	.120	.100	.086
Mean aerodynamic chord, ft	.923	1.077	1.238
Aspect ratio	2.860	2.620	2.420
Center line thickness ratio:			
6-X-67	.036	.030	.026
12-X-67	.072	.060	.051

Figure 4.- Geometry of wing-body combinations.

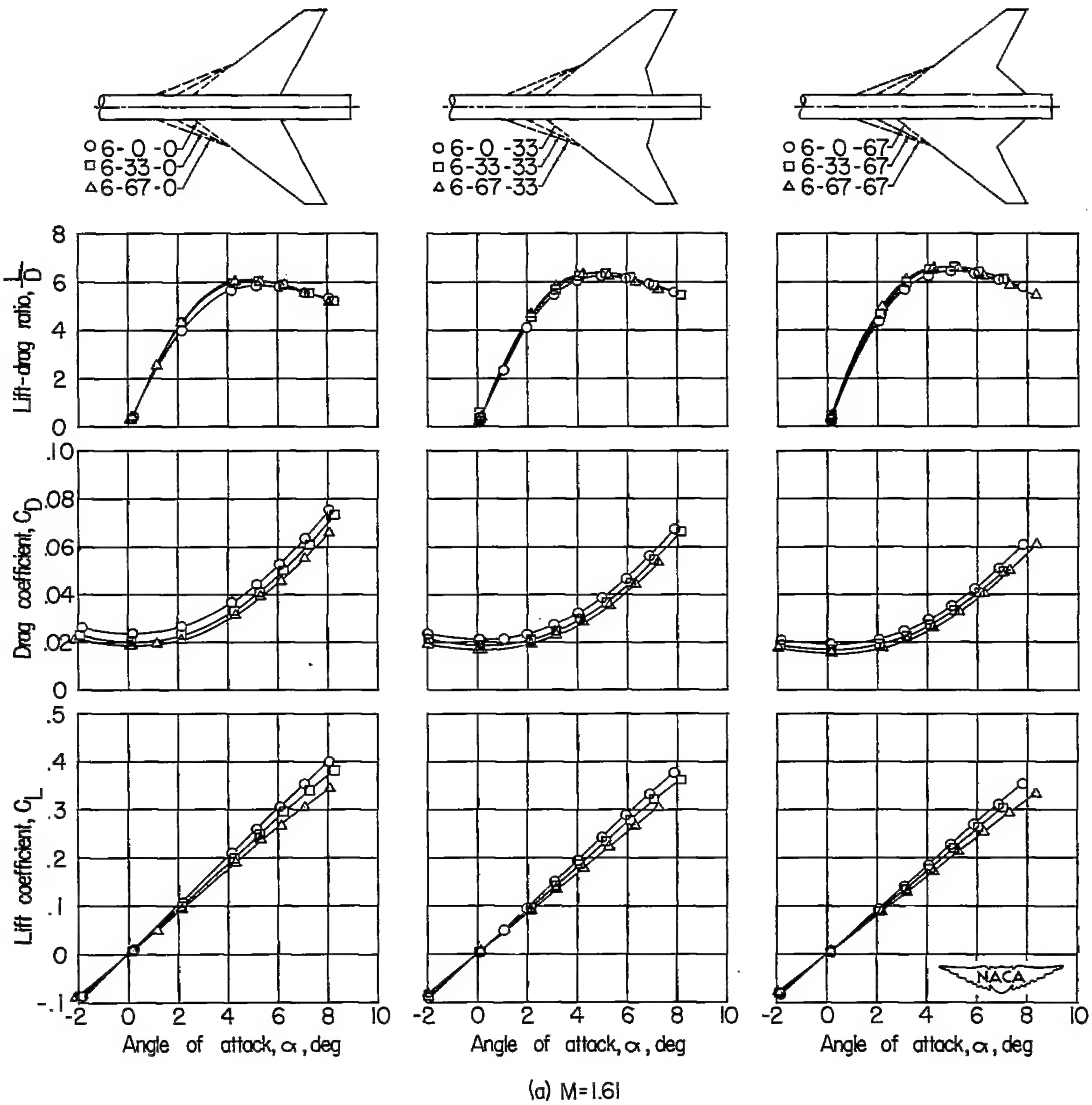
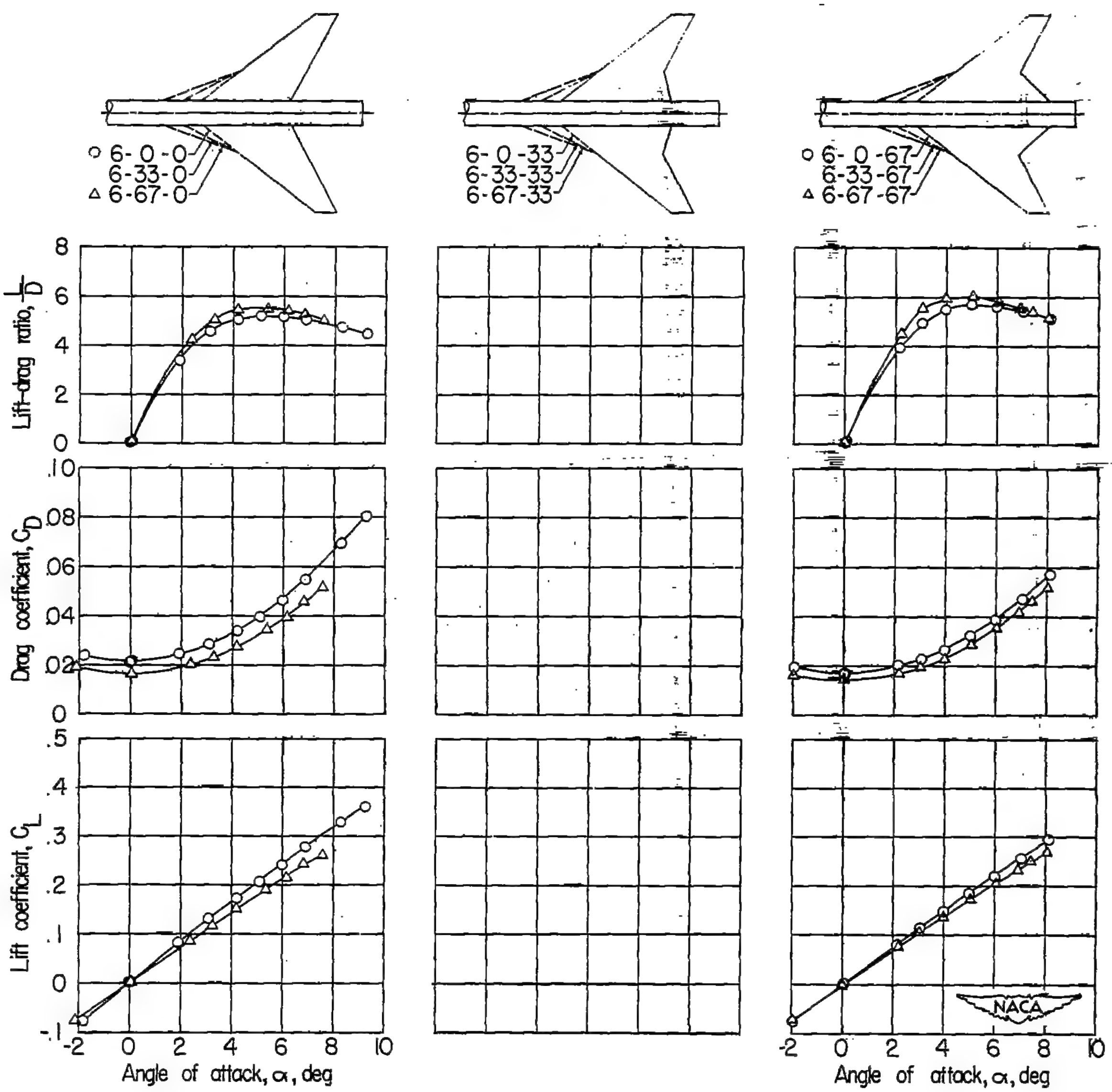
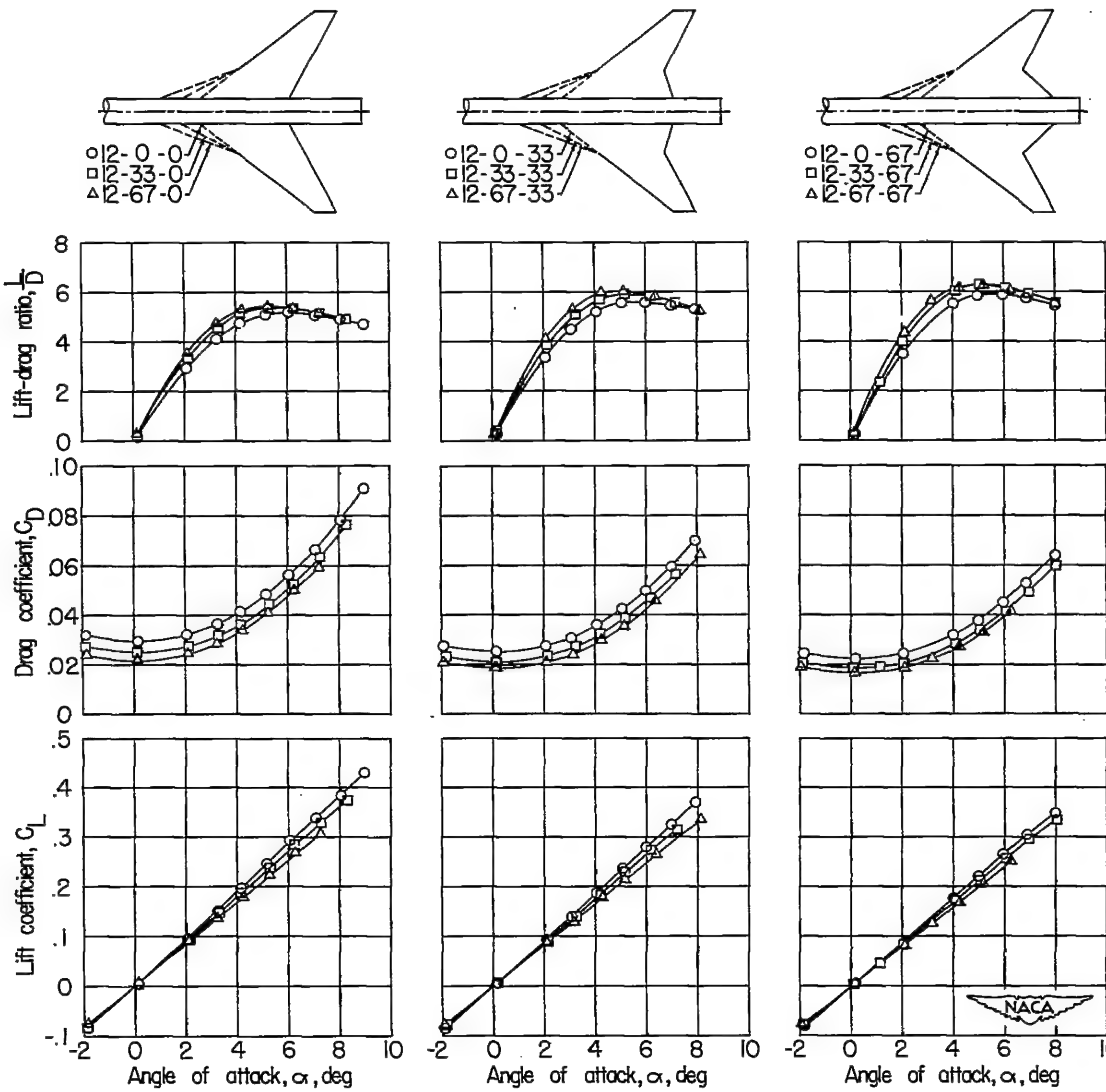


Figure 5.- Lift and drag characteristics for 6-X-X wing configurations.



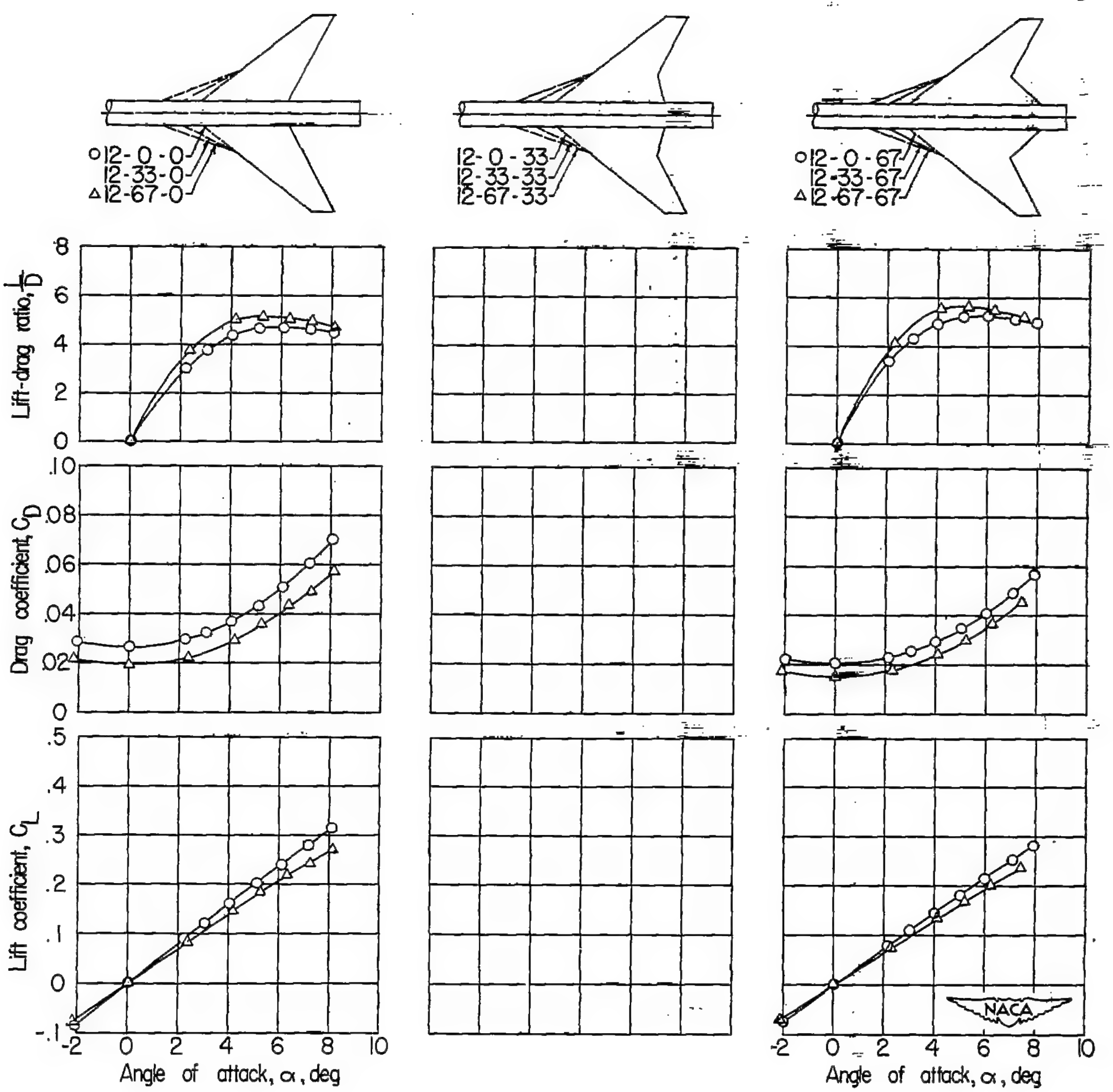
(b) M = 2.01

Figure 5.- Concluded.



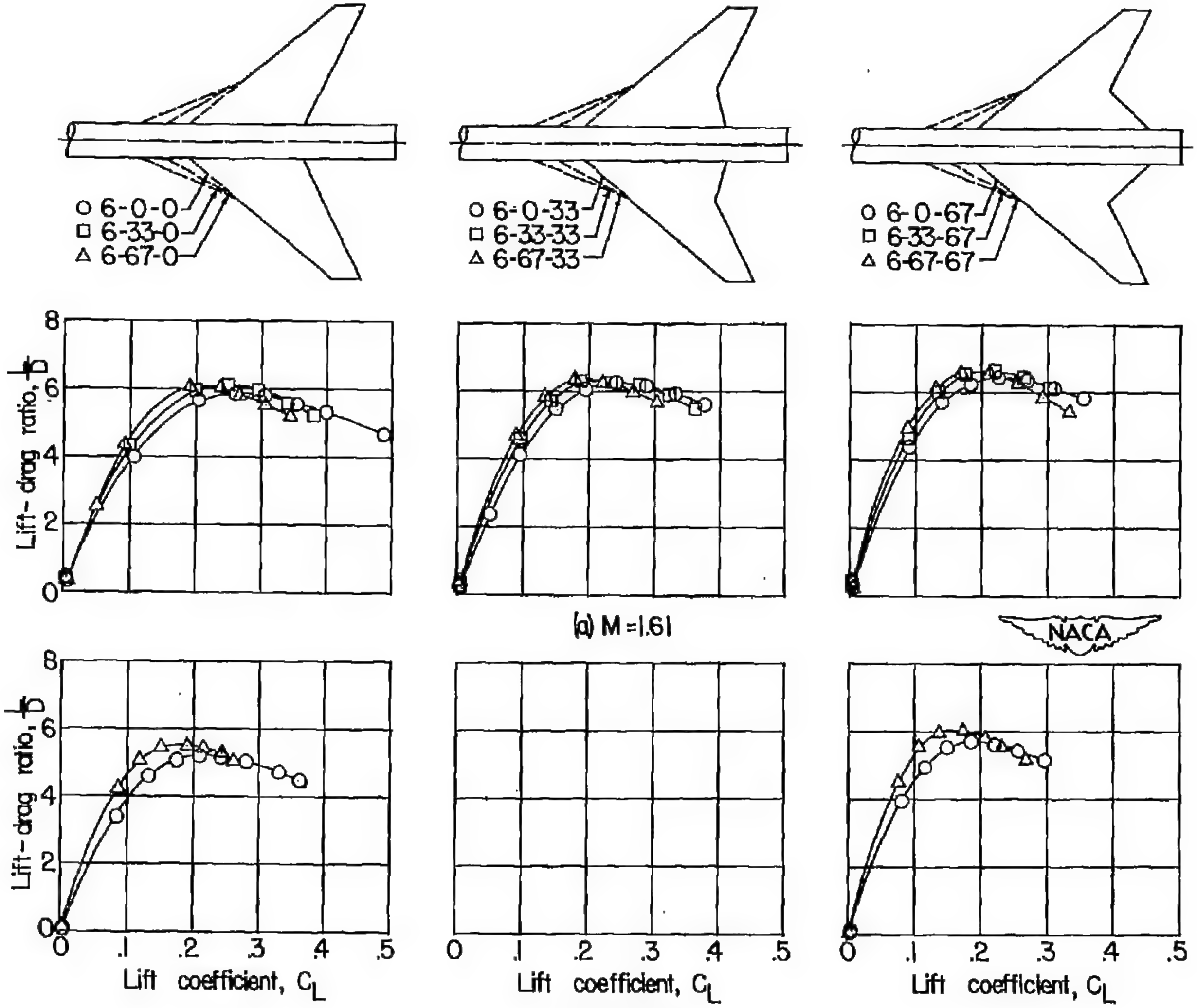
(a) M = 1.61

Figure 6.- Lift and drag characteristics for 12-X-X wing configurations.



(b) M=2.01

Figure 6.- Concluded.



(b) M = 2.01

Figure 7.- Lift-drag ratios for 6-X-X wing configurations.

CONFIDENTIAL

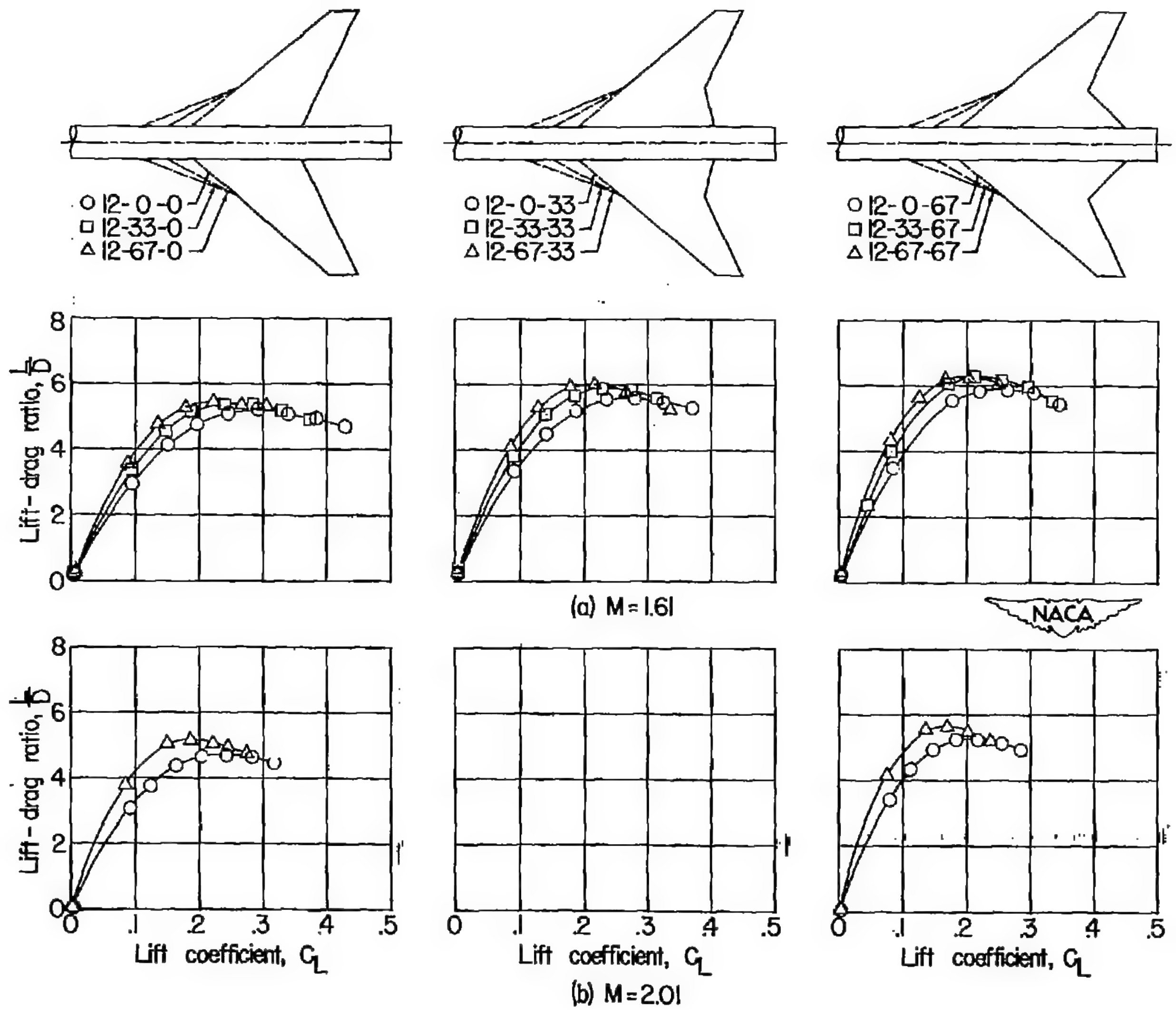


Figure 8.- Lift-drag ratios for 12-X-X wing configurations.

CONFIDENTIAL

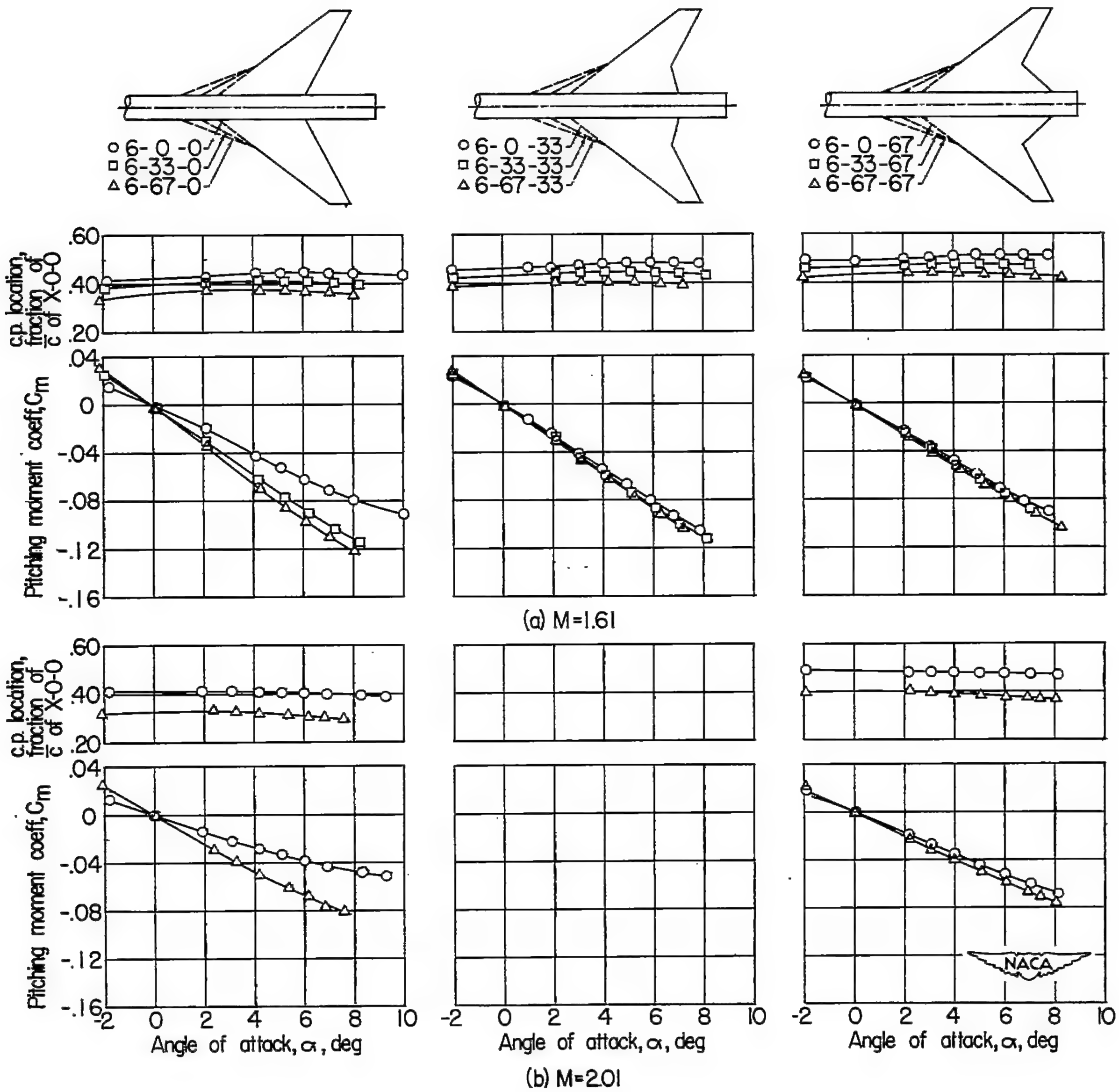


Figure 9.- Pitching moment and center-of-pressure characteristics for 6-X-X wing configurations.

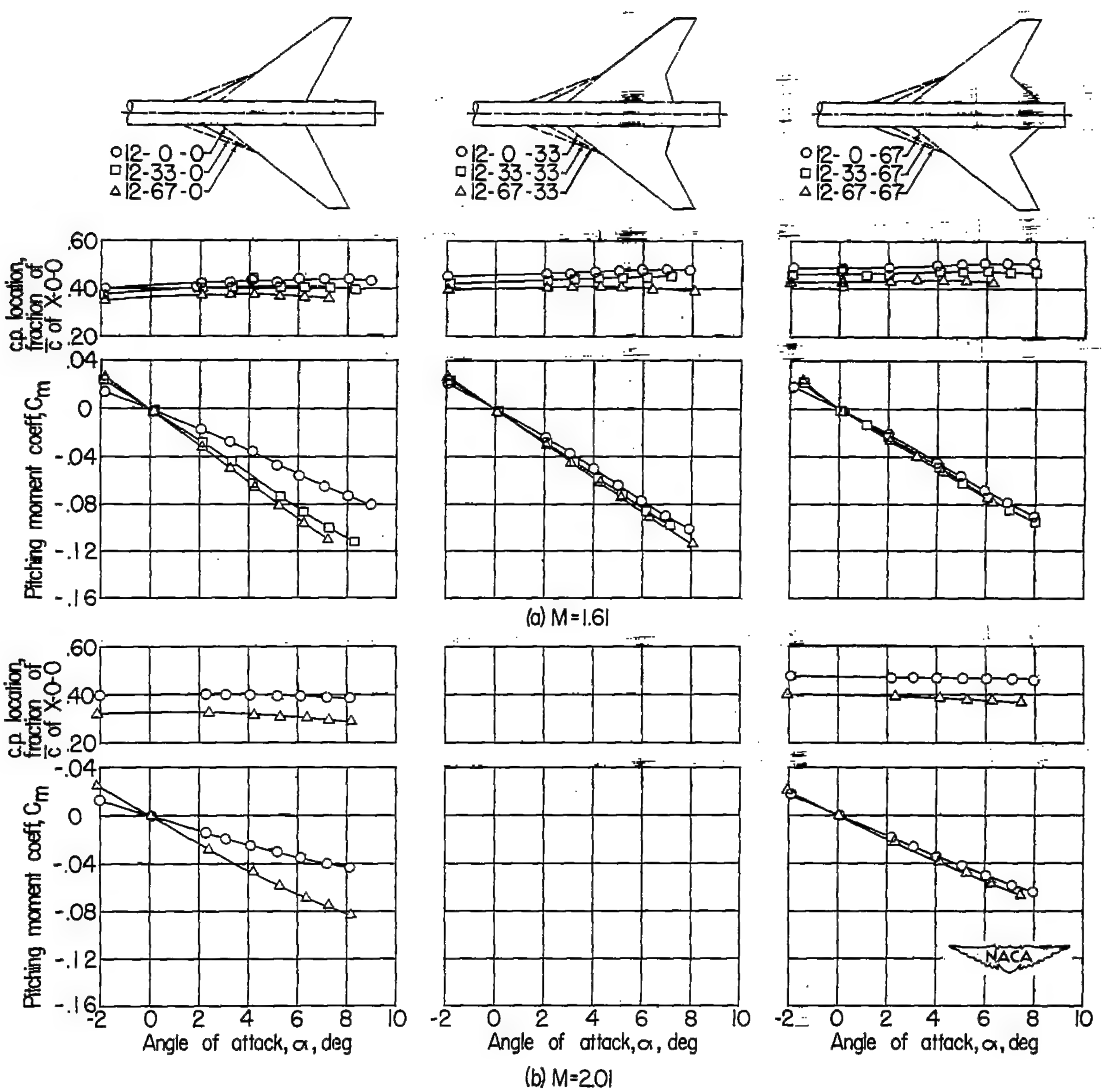
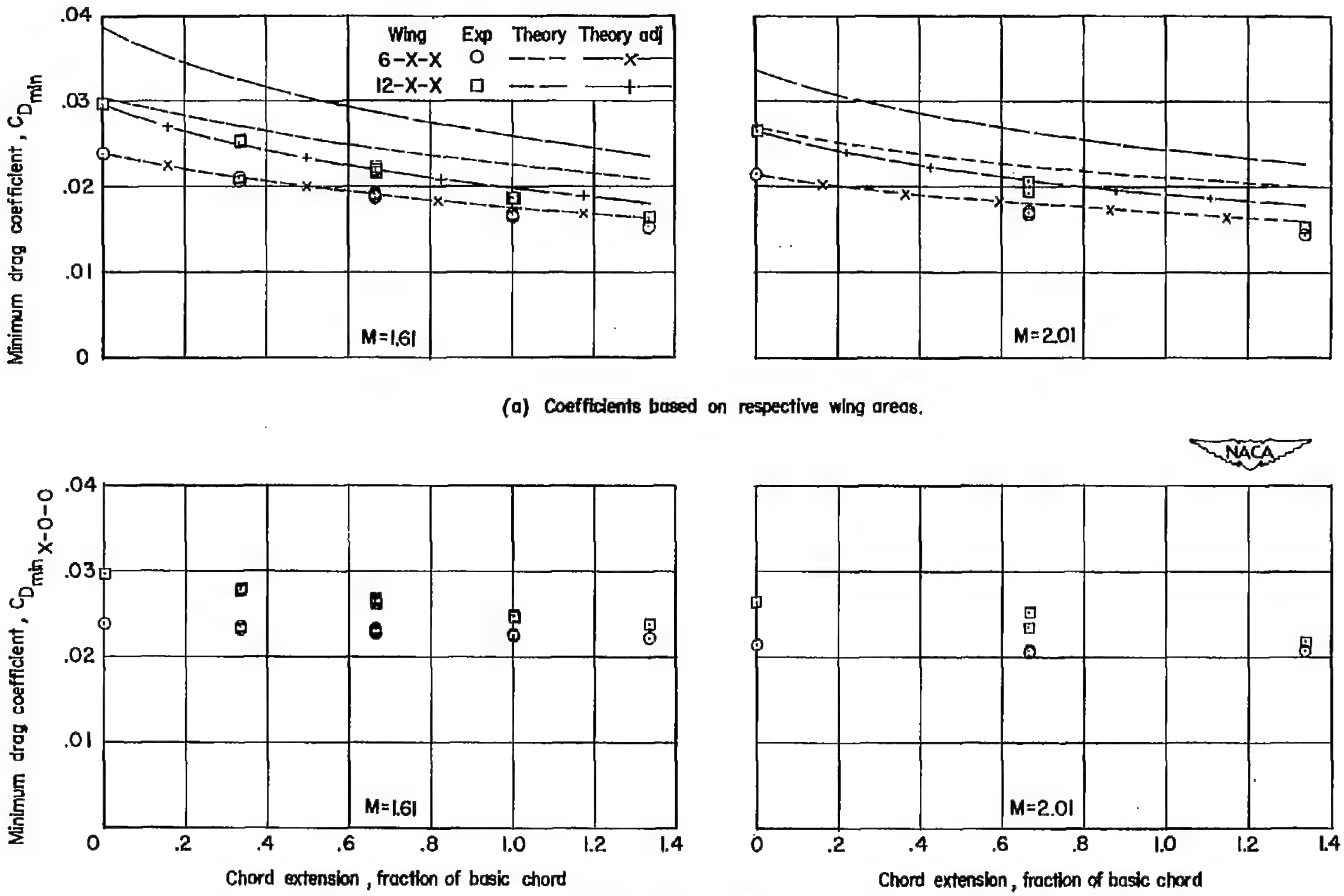


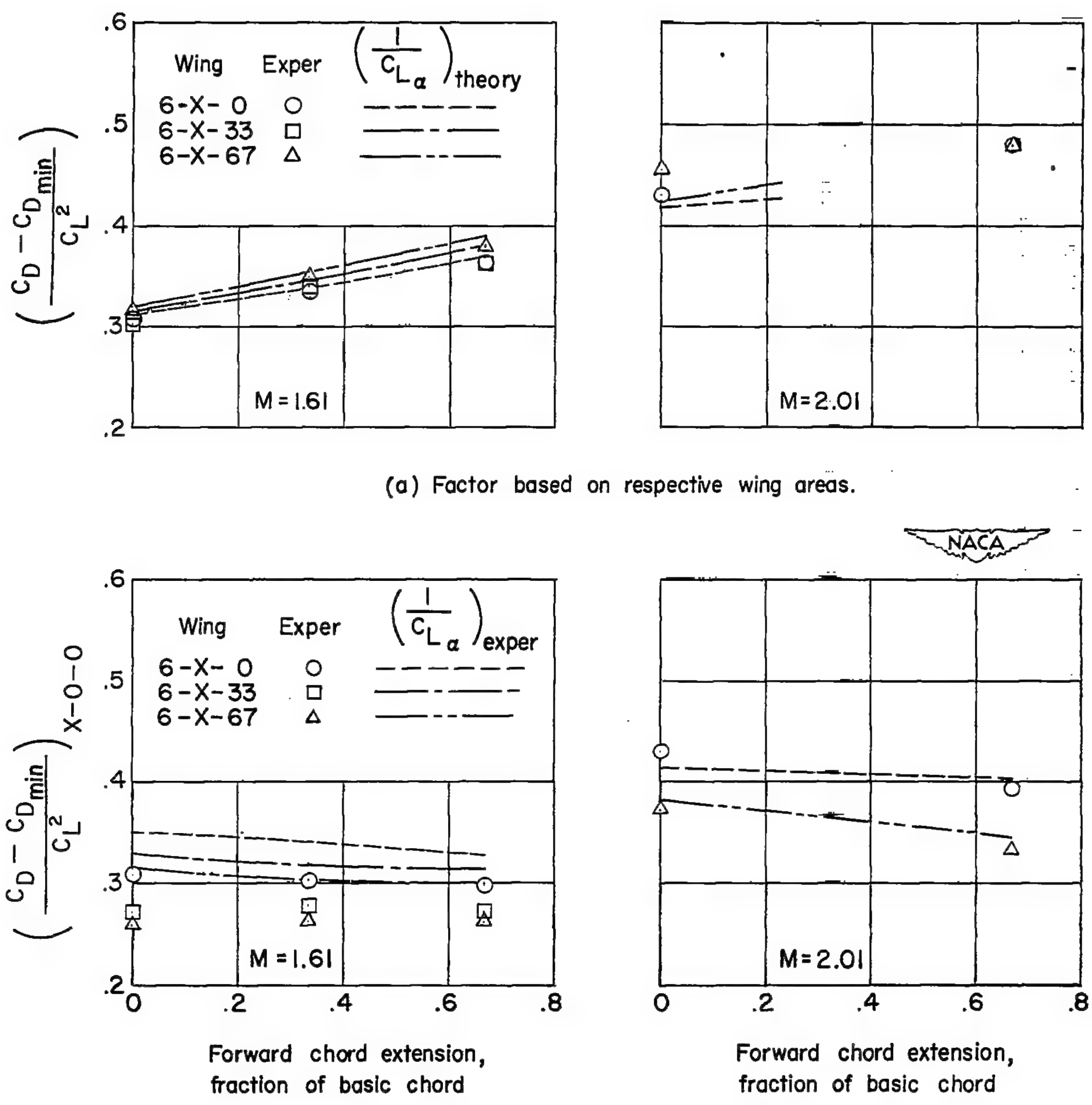
Figure 10.- Pitching moment and center-of-pressure characteristics for 12-X-X wing configurations.



(b) Coefficients based on area of X-0-0 wing.

Figure 11.- Minimum drag characteristics of 6-X-X and 12-X-X wing configurations.

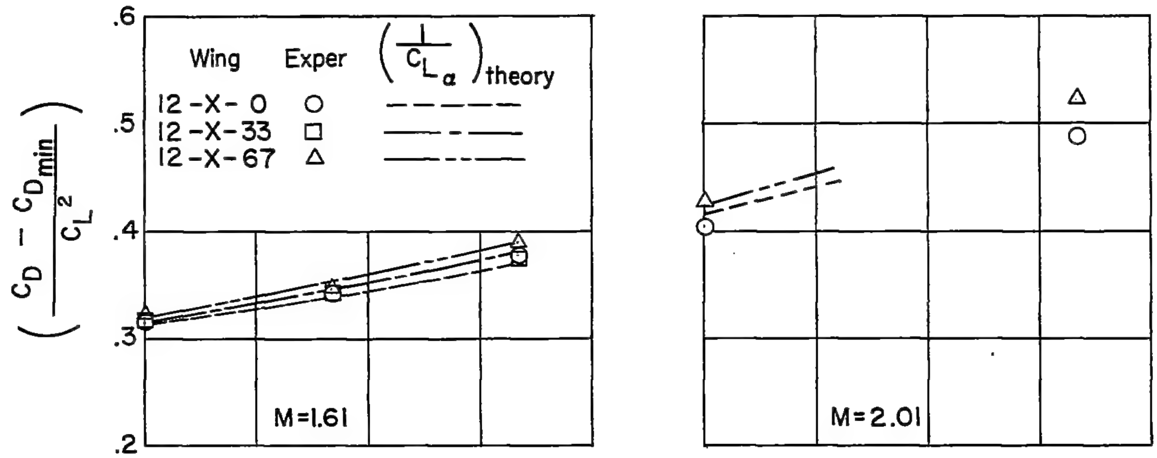
CONFIDENTIAL



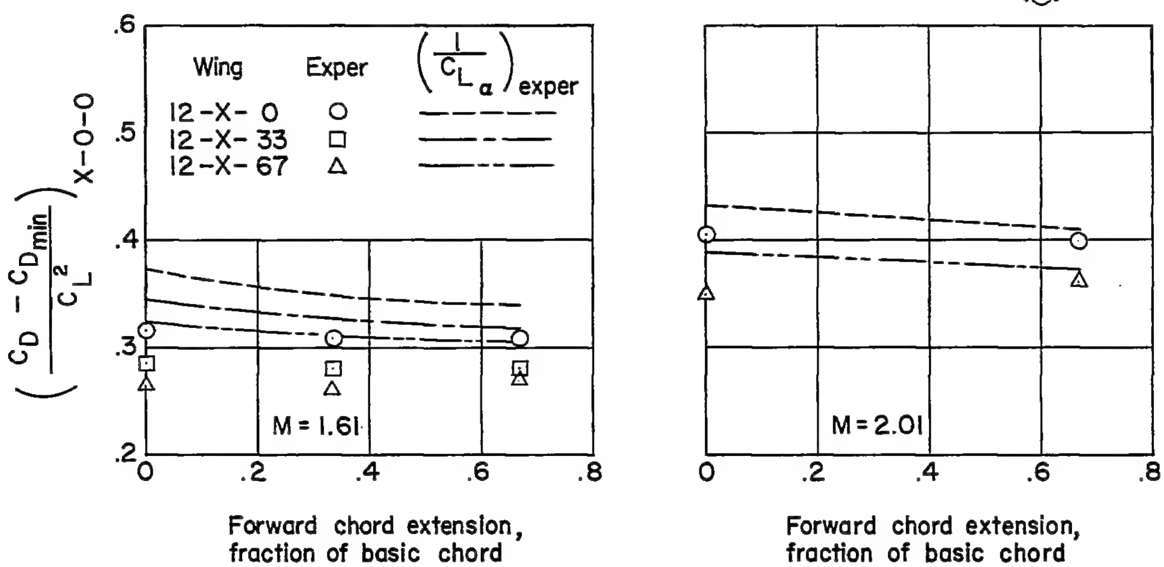
(a) Factor based on respective wing areas.

(b) Factor based on area of X-0-0 wing.

Figure 12.- Drag due to lift characteristics of 6-X-X wing configurations.

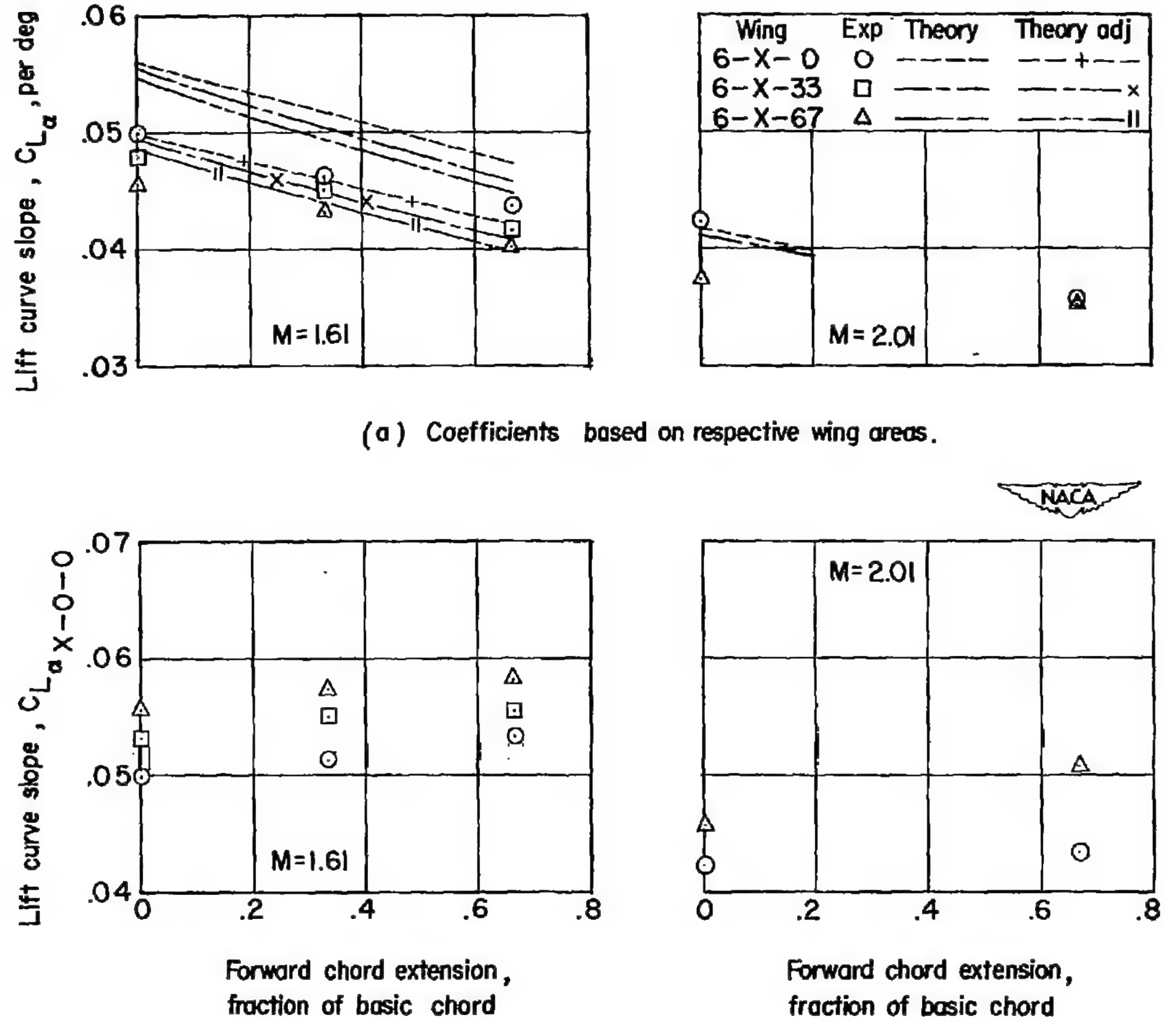


(a) Factor based on respective wing areas.



(b) Factor based on area of X-0-0 wing.

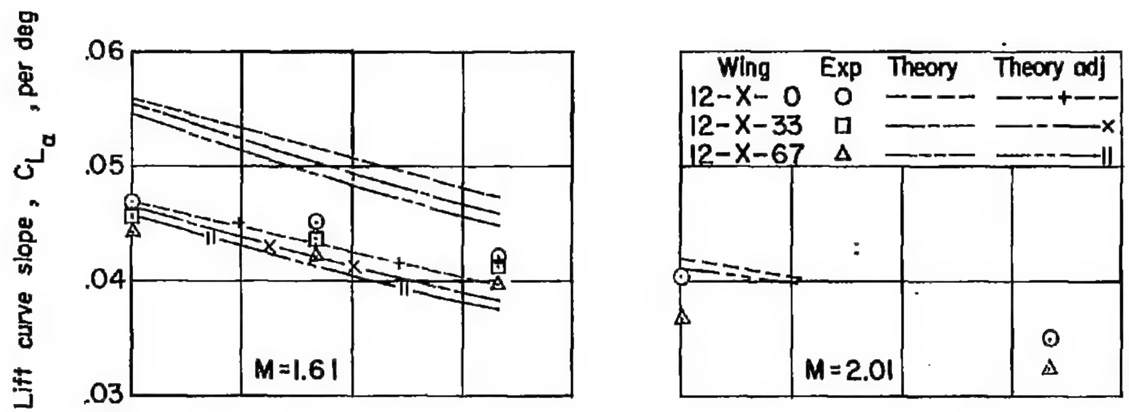
Figure 13.- Drag due to lift characteristics of 12-X-X wing configurations.



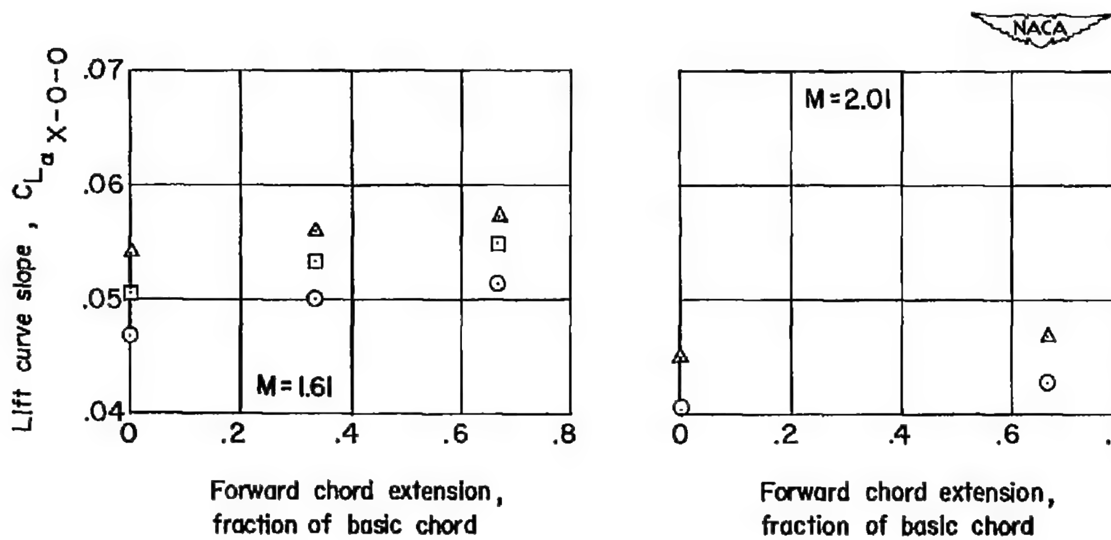
(a) Coefficients based on respective wing areas.

(b) Coefficients based on area of X-O-O wing

Figure 14.- Lift-curve slopes of 6-X-X wing configurations.



(a) Coefficients based on respective wing areas.



(b) Coefficients based on area of X-0-0 wing

Figure 15.- Lift-curve slopes of 12-X-X wing configurations.

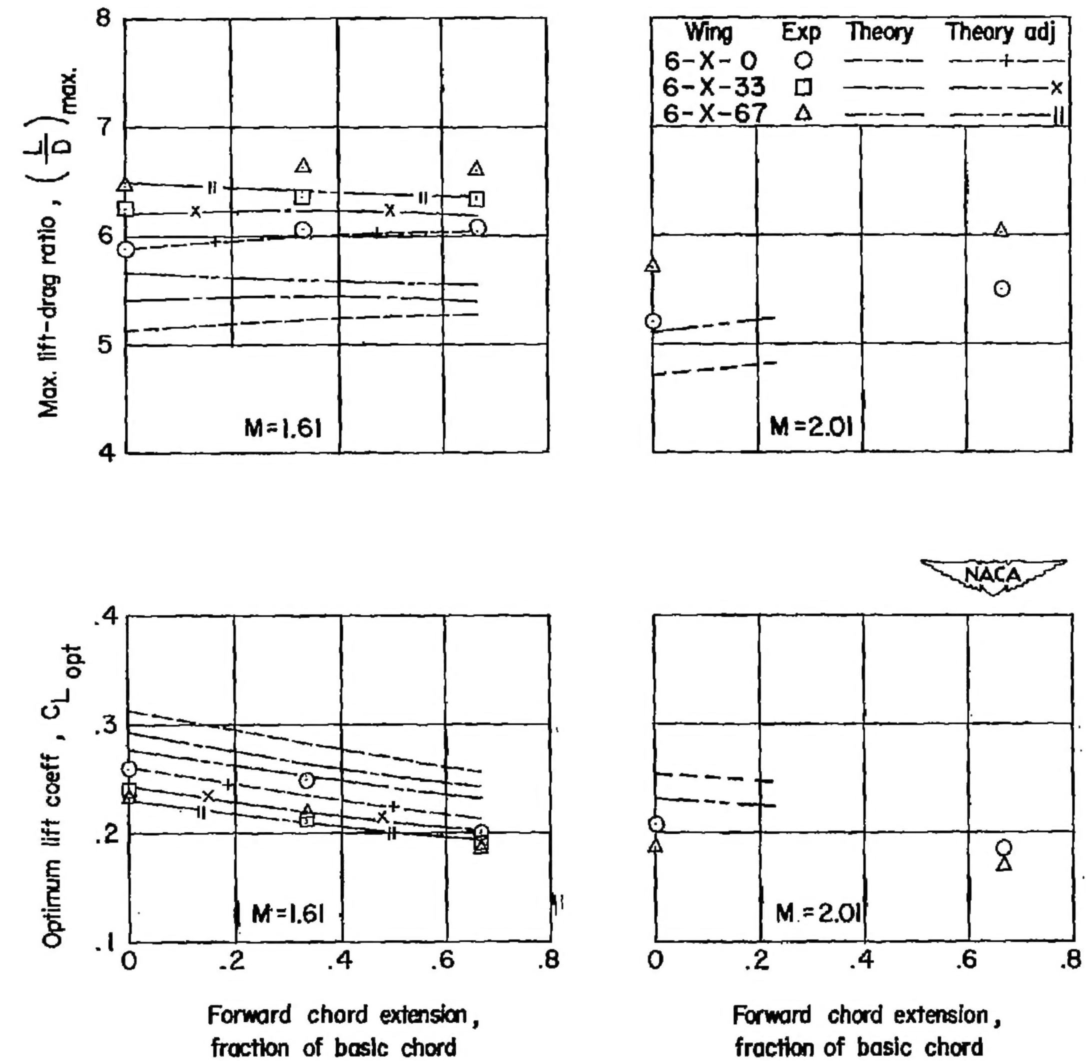


Figure 16.- Maximum lift-drag ratios and optimum lift coefficients for 6-X-X wing configurations.

CONFIDENTIAL

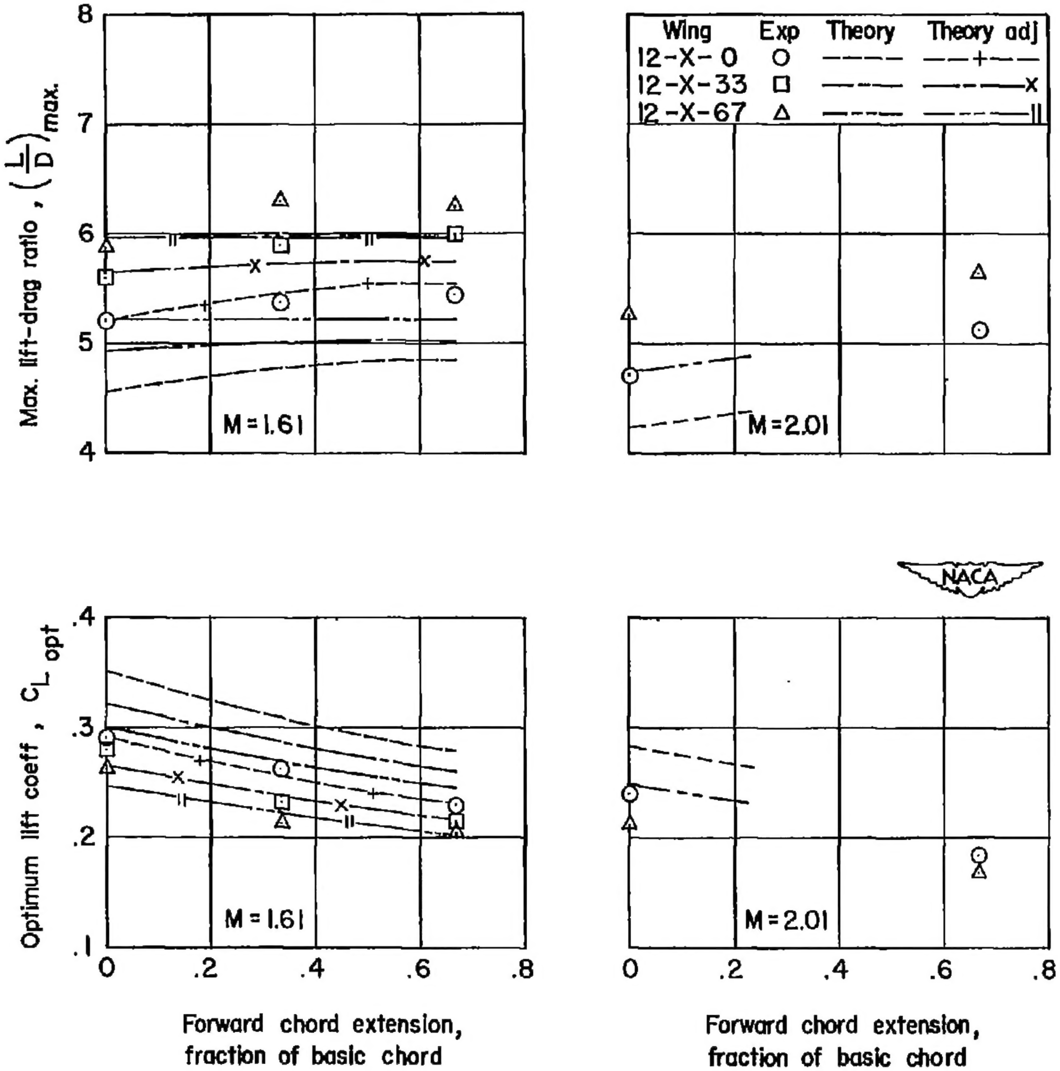


Figure 17.- Maximum lift-drag ratios and optimum lift coefficients for 12-X-X wing configurations.

CONFIDENTIAL

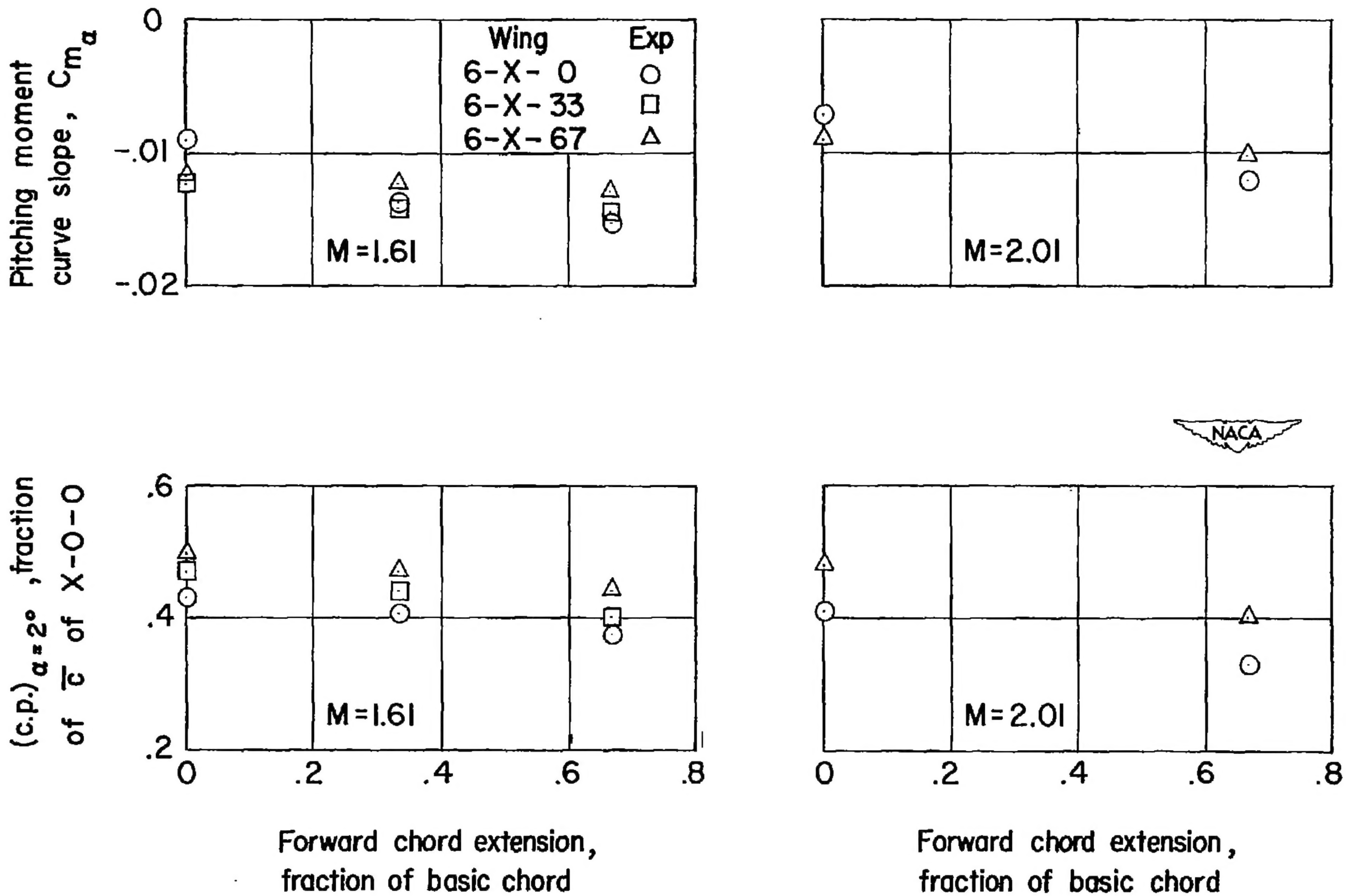


Figure 18.- Pitching-moment-curve slopes and representative center-of-pressure characteristics for 6-X-X wing configurations.

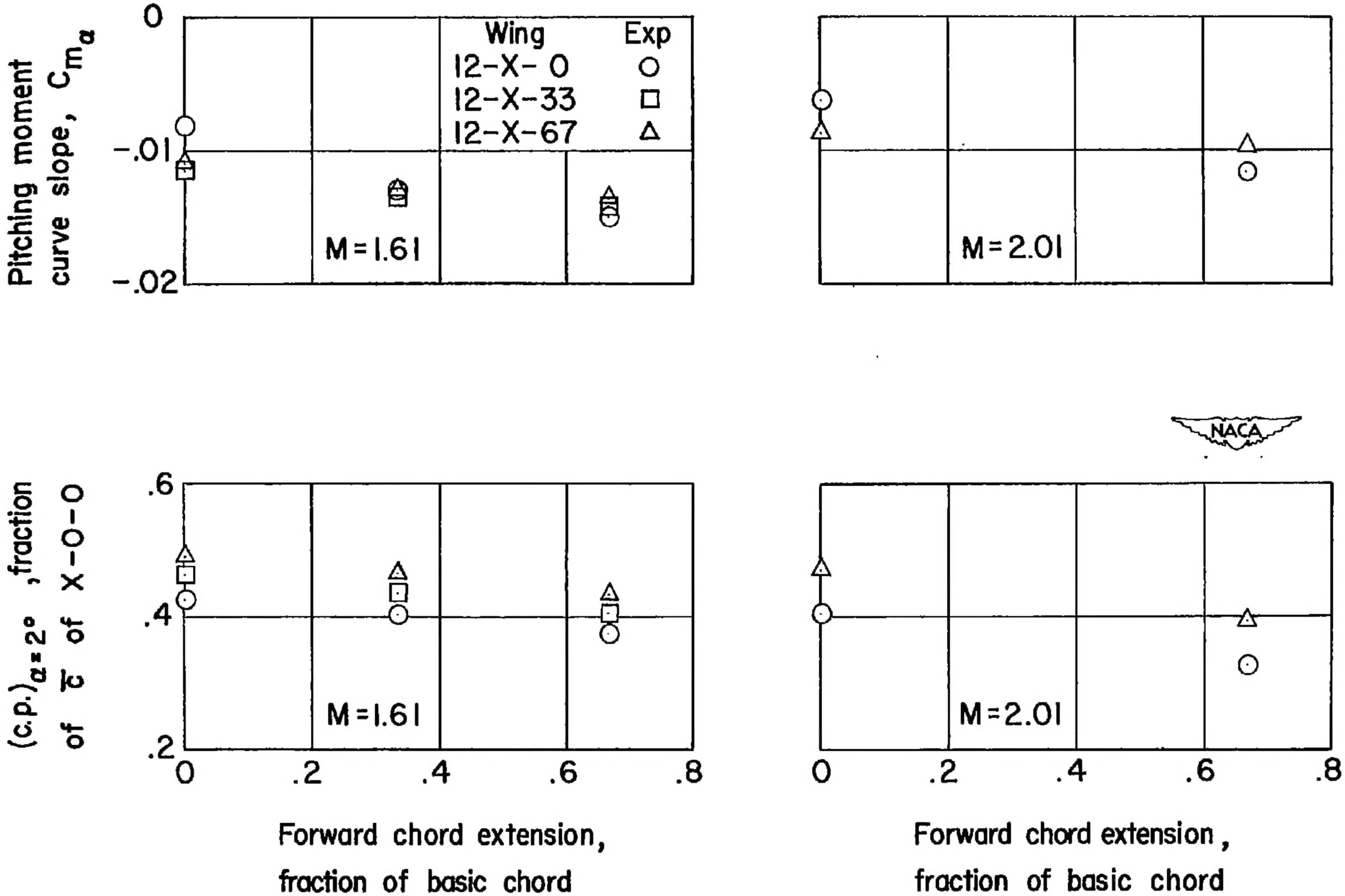


Figure 19.- Pitching-moment-curve slopes and representative center-of-pressure characteristics for 12-X-X wing configurations.